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Space Division
North American Rockwell

SATURN SUSTAINING ENGINEERING REPORT

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SPACE TUG POINT DESIGN STUDY FINAL REPORT
VOLUME II
OPERATIONS, PERFORMANCE & REQUIREMENTS

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Prepared for

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FOREWORD

The final report on the Tug Point Design Study was prepared by the North American Rockwell Corporation through its Space Division for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center in accordance with SA 2190 and Contract No. NAS 7-200.

The study effort described herein was conducted under the direction of NASA MSFC Study Leader, Mr. C. Gregg. The report was prepared by NR-SD, Seal Beach, California under the direction of Mr. T. M. Littman, Study Manager. The study results were developed during the period from 4 November 1971 through 11 February 1972 and the final report was submitted in February of 1972.

Valuable guidance and assistance was provided throughout the study by the following NASA/MSFC personnel:

C. Gregg - Study Leader
S. Denton - Structures
A. Willis - Avionics
J. Sanders - Propulsion
R. Nixon - Thermal Protection
A. Young - Flight Performance
R. L. Klan - Cost

The complete set of volumes comprising the report includes:

I Summary
II Operations, Performance, and Requirements
III Design Definition

Part 1 - Propulsion and Mechanical Subsystems, Avionic Subsystems, Thermal Control, and Electrical Power Subsystem

Part 2 - Insulation Subsystems, Meteoroid Protection, Structures, Mass Properties, Ground Support Equipment, Reliability, and Safety

IV Program Requirements
V Cost Analysis

This volume contains the detail analyses concerning flight and ground operations; vehicle flight performance and performance enhancement techniques; flight requirements; the basic design criteria; and the physical, functional and procedural interface requirements between the Tug and other systems.

ABSTRACT

The primary objective of the Tug Point Design Study was to verify through detail design and analysis the performance capability of a baseline design to deliver and retrieve payloads between 100 nautical miles/28.5 degrees inclination and geosynchronous. The Tug as groundruled for the study, is ground-based, reusable for 20 mission cycles, and is shuttled to and from low earth orbit by an Earth Orbital Shuttle (EOS) with a 65,000 pound payload capability. A 1976 state-of-the-art also was groundruled for the investigations.

The results of the effort show that the baseline concept can be designed to meet the target performance goals. Round trip payload capability to geosynchronous orbit is 3720 pounds; 720 pound margin over the established goal.

The design analysis performed to ascertain the Tug propellant mass fraction encompassed definition of the vehicle primary structure, thermal control, meteoroid protection, propulsion and mechanical subsystems, and avionics including power generation and distribution.

Graphite-epoxy composite material was determined to be feasible for Tug use and resulted in considerable weight savings. The concept of employing the primary load-carrying outer shell as a multi-function element integrating the meteoroid shield and insulation purge bag requirements is also feasible and enhances design simplicity. In addition, the use of a dual-mode pressure schedule during boost to orbit when applied loads are highest resulted in minimum tank weight. This, combined with an integrated gaseous O_2/H_2 auxiliary propulsion for stability and control, main tanks prepressurization, and fuel cell usage yield a minimum weight and operationally simple system.

Reliability and Safety analyses verified that no single failure of a component would result in a critical or unsafe condition. This was accomplished employing redundancy as required, notably in propulsion subsystems valving and attitude control components.

Program requirements were developed to verify the feasibility, producibility and operational capability of the point design. The results indicate that an "on-condition" maintenance approach similar to that used by commercial airlines and military operations would effectively serve Tug requirements.

Technology development study effort was concentrated on identifying the technologies needed for the baseline design. The more critical technologies requiring development include high performance engines, high performance insulation, large composite structures, and avionics.

A preliminary program development schedule was structured summarizing the integrated activities necessary to support the Tug through design development, production, and ground and flight testing.

The cost analysis performed covered the five major cost categories of DDT&E, first unit production, SR&T, average flight maintenance and refurbishment, and flight test vehicle refurbishment.

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1.0 INTRODUCTION

The Space Tug is a high performance propulsion stage designed to operate as a third stage for the two-stage Space Shuttle. Because of the nature of the Tug mission, performance capability is very sensitive to Tug mass fraction. This point design study, the results of which are presented in this final report, was conducted to answer the questions "What Tug mass fractions are really achievable by 1980?", and "What level of technology effort is required in order to build a Tug having the high performance defined in NASA/MSFC's Study Plan (Reference 1)?". Both questions are discussed below.

1.1 BACKGROUND

Several pre-Phase A Tug/OOS* studies have been conducted for NASA and USAF agencies with a wide variation in the mass fractions quoted. NR performed a reusable Space Tug study for NASA-MSC in 1970-71 (Reference 2) and both NR and MDAC evaluated OOS feasibility for SAMSO/Aerospace Corporation in 1971 (References 3, 4). Additionally, two European teams conducted Tug system studies for the European Space Agency (ELDO) during 1970-71 (References 5, 6). In-house investigations also have been accomplished by MSFC and Aerospace Corporation. These studies considered a wide variety of design concepts and autonomy limits, ground and space-based operational requirements, degree of reusability, unmanned and manned payload implications, single and multi-stages, and different technology bases.

Projected NASA and DOD missions for the 1980's and beyond demand a Tug designed for a high degree of reusability and operational flexibility to assure significant improvement in space flight economy. Furthermore, Tug design must be compatible with Shuttle cargo bay size, weight limitations, and environment. For a ground-based system, consideration also must be given to Shuttle transport of a mated Tug/Payload.

1.2 OBJECTIVES

This point design study had one primary aim which was to be verified by design detail and analysis; namely, that a reusable, ground-based Space Tug with an IOC target by about the end of 1979 (1976 state-of-the-art) can carry a 3000-pound round trip payload between orbits at 100 nautical miles/28.5 degrees and geosynchronous. The key constraint was use of a Space Shuttle having a 65K pound orbital delivery capability. A minimum usable propellant mass fraction of 0.895 also was desired. Additional study objectives were to (1) define the necessary supporting research and technology (SR&T) activities and their associated funding, and (2) determine Tug development, first production, and maintenance/repair costs.

*Orbit-to-Orbit Shuttle

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1.3 STUDY SCOPE

Due to study time limitations, the detail design of an integrated system was performed only for a baseline concept. The concept was derived from MSFC's Study Plan and NR-selected materials, fabrication techniques, and subsystems resulting from currently available data and new trade studies.

Concurrent with the baseline study, options were evaluated having the potential for improving Tug mass fraction and mission performance. Emphasis was placed on the areas of alternate materials and subsystems, flight mode and operational variations, and use of advanced technology.

The study logic of Figure 1.3-1 depicts the major functional activities and outputs of these activities. The analyses performed to satisfy study objectives can be subdivided into three inter-related major efforts which started at study outset and ran concurrently to completion. Initiation of these efforts at the same time was made possible by the large amount of technical data available from the data base indicated. System requirements and criteria definition and program support gave the design definition effort the input data necessary for realistic structural, mechanical, thermal, and avionics subsystems design taking into account reliability and safety requirements. The three major tasks formed an iterative loop to the extent that the short study schedule permitted. As the design of each component and subsystems evolved, it fed the results back to the supporting activities which served to increase the depth of analysis and visibility of the overall system characteristics with each succeeding step. This approach also lent itself to the timely establishment of performance sensitivities and development of potentially attractive subsystem concepts.

1.4 STUDY GUIDELINES

This section highlights those elements of the NASA Study Plan (Reference 1) which were most influential in directing the NR effort toward the achievement of the aforementioned objectives.

Key Assumptions and Guidelines

The items listed below *provided* the key design and operational drives for the Tug:

1. 1976 Materials & Concepts Technology

2. Unmanned Design, Fail-Safe Operation

3. Reusable - Lifetime of 20 Missions

4. Ground Based - Refurbishment After Each Mission

5. 6-Day on Orbit Stay Time Unattached to Shuttle

6. Flt Between 100 NMI Circular, 28.5° Inclin & Geosync

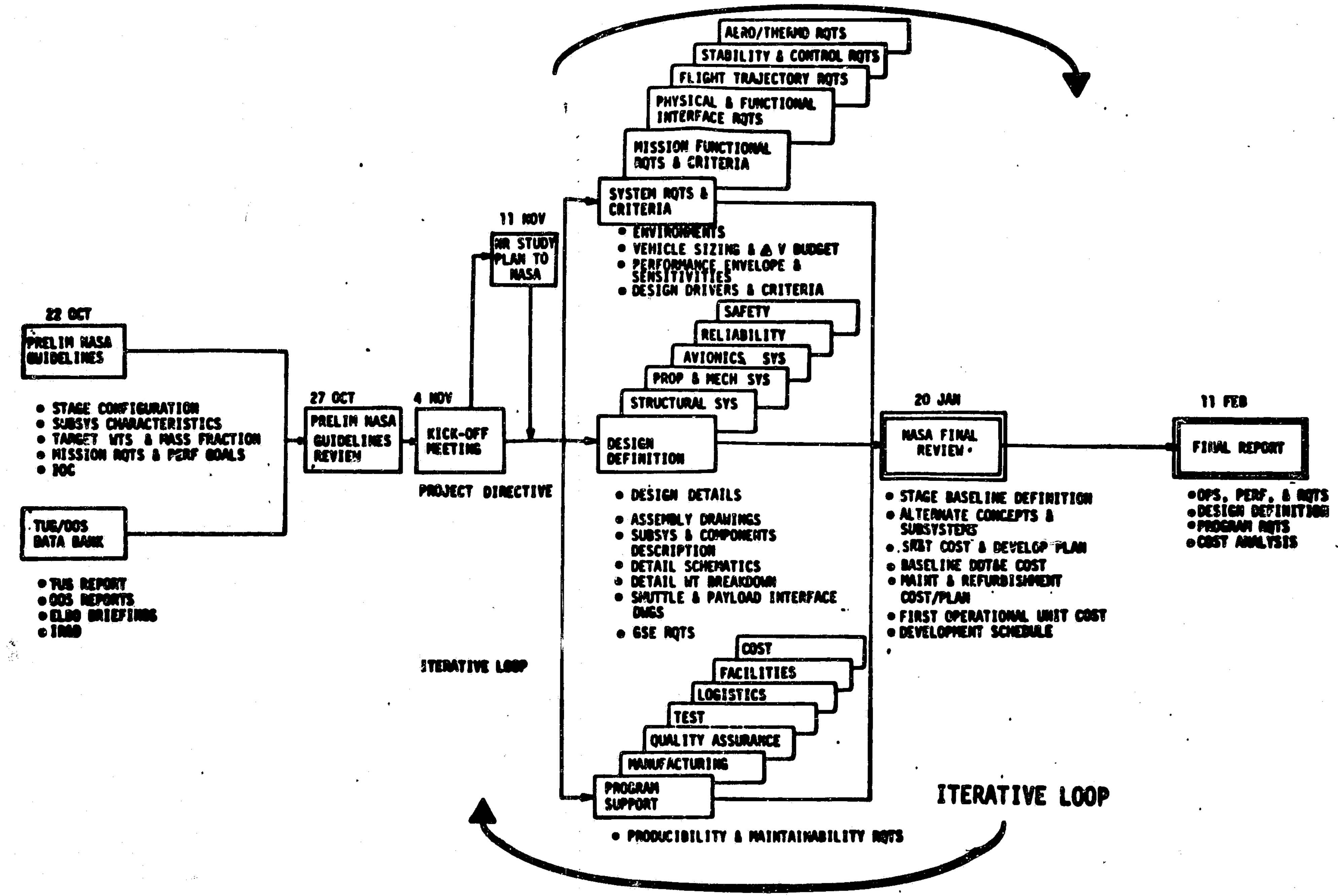

Figure 1.3-1 Top Level Study Logic

7. Payload Deliver/Retrieve Mixes

Baseline	3K/3K	
Alternates	0/4.16K 8.06K/0	Sizes Outer Shell Structure

8. Abort From Orbit Only & Propellant Dump/Inerting From Cargo Bay

9. Integrated MPS & APS Subsystems

Low vehicle weight was a key design criterion due to the aforementioned performance objectives. Therefore, strong emphasis was given to the use of advanced materials and concepts deemed part of the 1976 technology base, but achieveable without incurring severe cost penalties or high development risks. Fail-safe (FS) operations also provide for lower weight due to redundancy limitations (compared to the more demanding FO/FS requirements as employed in the OOS studies). However, FS does necessitate the highest practical component reliability to achieve an acceptable (over 0.9) mission success probability. Fail-safe is defined here as no failure modes which would cause an unsafe situation for the Shuttle or its crew, or the Tug's payload to be destroyed. In the event of mission abort (limited to abort from orbit) while the Tug is still in the cargo bay, propellant dumping, tank inerting, and subsystems safing are required. These capabilities also are specified for normal re-entry and landing conditions to minimize hazards.

Unmanned design necessitates a high degree of subsystem/operational autonomy with ground support provided as emergency backup or when it yields weight and design simplicity advantages.

Reusability for 20 mission cycles (which may cover a period in excess of 3 years) can only be achieved in a practical cost-effective sense if airline-type servicing techniques are developed for Tug (as is planned for Shuttle). Strong attention must be given to assure a design compatible with this approach (accessibility, ease of inspection, and checkout).

The six-day orbital stay time affects cryogenic tankage protection and the total space exposure (for 20 missions) specifies meteoroid shielding requirements.

The baseline (3K round trip) payload capability represents the most demanding from a performance (mass fraction) viewpoint. However, normal Shuttle ascent and descent carrying the Tug and the alternate payloads were employed to size the Tug outer shell structure, based on the flight load factors provided by MSFC for the study.

One additional assumption agreed to between MSFC and NR, use of an integrated LO_2/LH_2 propellant system for both main and auxiliary propulsion, provides design simplicity as well as weight and performance advantages.

Tug Baseline Concept

The NASA baseline configuration (Figure 1.4-1) which served as the starting point for this study is a single stage orbital propulsion system. It is limited to a maximum overall diameter of 15 feet and a maximum length of 35 feet, including Shuttle/Tug and Payload/Tug docking mechanisms. This system is intended to separate from the Shuttle orbiter at 100 n mi/28.5 degrees with a 3000-pound payload (15 ft x 25 ft) attached, ascent to geosynchronous orbit, deploy the up payload, retrieve a 3000-pound payload within 6000 n mi of the deployed payload, return to the near-vicinity of the Shuttle, redock, and return to earth. Payload center-of-gravity was defined as being at the geometric center of the 15 x 25 payload envelope.

The Tug has a non-integral tankage arrangement and is sized for a total propellant capacity of 56,394 pounds including 350 pounds of reserve plus allocations for reaction control/auxiliary propulsion (APS), fuel cell, residuals, and losses. The LH_2 tank has hemispherical bulkheads and a cylindrical section, whereas the LO_2 tank consists of two ellipsoidal bulkheads.

The docking systems are designed such that the active portion is left with the Tug in the Tug/payload interface and with the Shuttle in the Tug/Shuttle interface.

Other pertinent features are indicated on the profile. It should be noted that the Tug is attached at its aft end to the forward part of the orbiter cargo bay and thus is transported between Earth and orbit in an inverted attitude.

Tug Weight Targets

Table 1.4-1 lists the "bogey" weights provided by MSFC as design goals to assure meeting mass fraction requirements with the constraints of a 65K Shuttle capability and a 3K Tug payload. No specific allocation was made for Tug-supportive hardware and fluids which remain in the EOS cargo bay. Instead, these were assumed to be contained within structure and other subsystems.

Structure includes all dry structure (docking mechanisms, meteoroid shield, outer shell, supports, thrust structure) and tankage subsystems. Thermal control includes cryogenic insulation, avionics cooling/heating hardware, and purge systems. Avionics contains GN&C, communications, data management, on-board checkout, power generation and distribution, rendezvous and docking, and Tug electrical interfaces for ground and Shuttle. Propulsion includes dry main engine, propellant feed, pressurization, fill/drain and vent/purge umbilicals, propellant dump, tank baffles/screens, APS thrusters/feed system/tanks, main engine actuator, and ullage control.

Non-usable fluids include propellant reserves, pressurant, thermal control fluids, and residuals. The main engine propellant bogey weight contains all propellant burned by the main engine during a nominal mission. APS propellant includes all burned attitude control and small delta-V translational maneuver

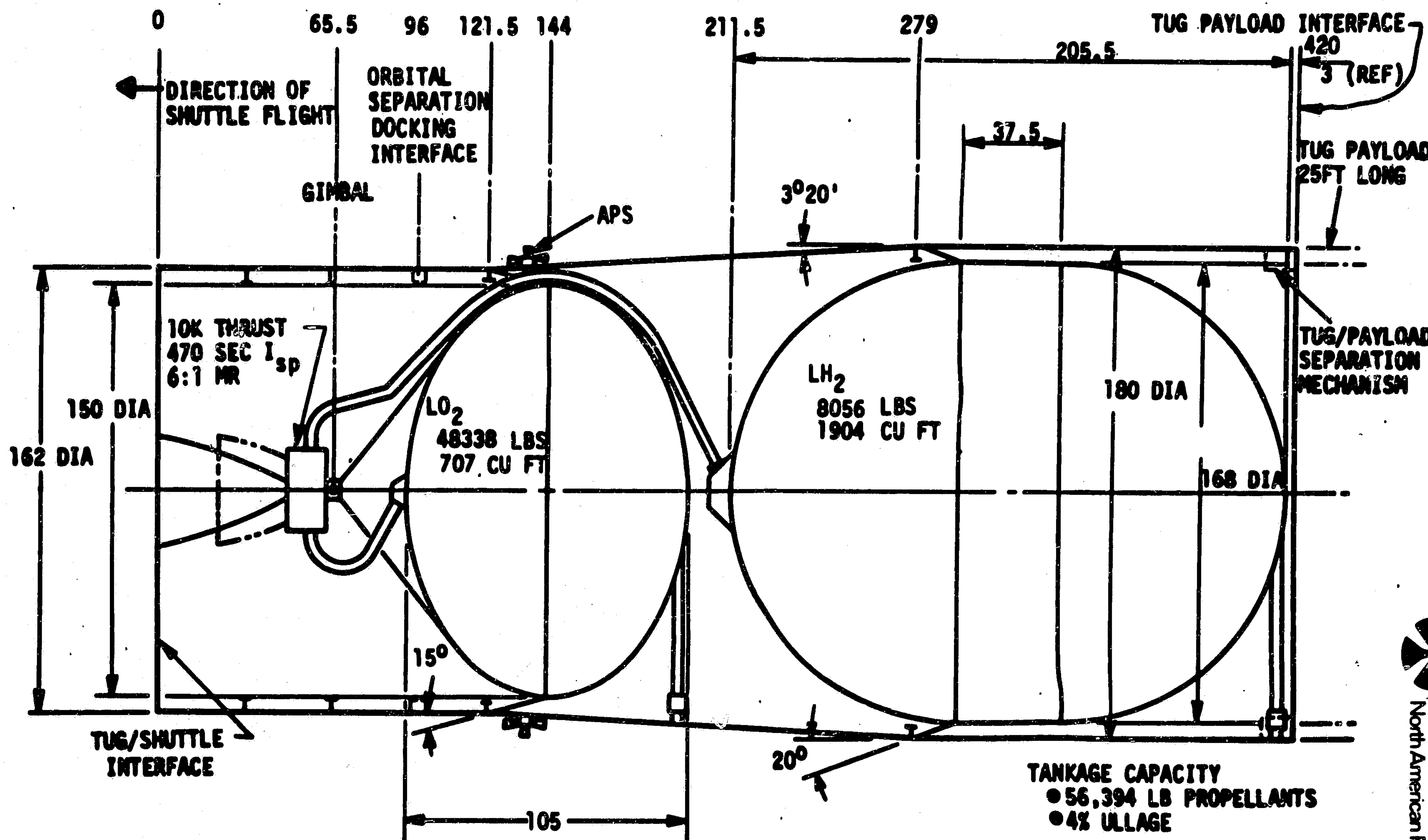

Figure 1.4-1 NASA Tug Baseline Concept & Sizing

requirements during a nominal mission. The miscellaneous fluids category contains all other unburned fluids (fuel cell, reactants, and vent/childown/start-stop losses). These have been numerically lumped together with non-usable fluids in the bogey weight table.

TABLE 1.4-1. TUG BOGEY WEIGHTS

	WEIGHT (LB)
STRUCTURE	2,552
THERMAL CONTROL	476
AVIONICS	1,011
PROPULSION	1,057
DRY WEIGHT	5,096
10% CONTINGENCY	510
NON-USABLE FLUIDS	842*
BURNOUT WEIGHT	6,448
USABLE MAIN ENGINE PROPELLANT (Wp)	55,148
USABLE APS PROPELLANT	404
MISC FLUIDS & LOSSES	--
TUG FLIGHT WT AT TUG/EOS SEPARATION (W_G)	62,000
EOS PAYLOAD - CHARGEABLE INTERFACE PROV	--
TUG GROSS WT AT EOS LIFTOFF	62,000
GROSS EOS PAYLOAD AT LIFTOFF	65,000

MASS FRACTION, $\lambda = \frac{W_p}{W_G}$

*INCL 350 LB PROP RESERVE

2.0 OPERATIONS

2.1 LIFE CYCLE FUNCTIONS

A primary objective of the Tug Study was to develop a feasible point design for an orbit-to-orbit Shuttle which, as the third Stage of the Space Transportation System (STS), is to place and/or retrieve payloads at various inclination in earth orbits including geosynchronous orbit. The Tug defined in this study is designed to be carried to the 100 N.Mi. 28.5° inclined operations orbit by the Space Shuttle Orbiter. In this orbit the Tug and payload is deployed from the Shuttle Orbiter and the Tug delivers the payload to its desired orbital location, retrieves a second payload, and returns to the Shuttle Orbiter in the operations orbit (100 N.Mi, 28.5° inclination). The Shuttle Orbiter docks with the Tug and the Tug is brought back to earth in the Shuttle Orbiter cargo bay. A key step in assessing the relationship of the Tug to the STS development and operational cycle is an identification of top level functions that would influence Tug design and operation.

In examining the top-level functions ascribed to the Space Transportation System, 14 major functional blocks were identified in the cycle of production, use, and reuse of STS elements. A functional flow block diagram (FFBD) showing the relationship of these top-level blocks (FFB) are presented in Figure 2.1-1. The recovery function is shown separately from mission flight operations (since it occurs on the ground) to present transportation as an alternative to ferrying, especially for the Tug (and its payload on retrieval missions), and to separate the assembly function from the mission oriented function of preflight checkout and launch. With these functional categories, it is possible at the top level to isolate ground operations involved in inspection, maintenance, storage, handling, repair and assembly of the Tug from the ground-based operations involved in mission control and support.

The initial-phase of the development and operation cycle is, in the main, concerned with the three functions relating to recruitment of personnel, manufacture of system components, and development of facilities (FFB's 11.0, 12.0, and 13.0). These activities are followed by a functional block involving the assembly, testing, and acceptance of system elements (FFB 10.0) and an intermittent but continuing function of maintaining the facilities developed for the program (FFB 14.0). These five functions are not considered in detail as they were deemed to have little, if any, influence on Tug design requirements or operational timelines.

Transportation of STS elements from the producing plant to the assembly/launch facility (FFB 9.0) follows the acceptance function. The transport block is followed by a functional block covering the maintenance, repair, assembly, and readying of the systems for the mission phase operations (FFB 8.0). It is at this point that reuse of system elements begins;

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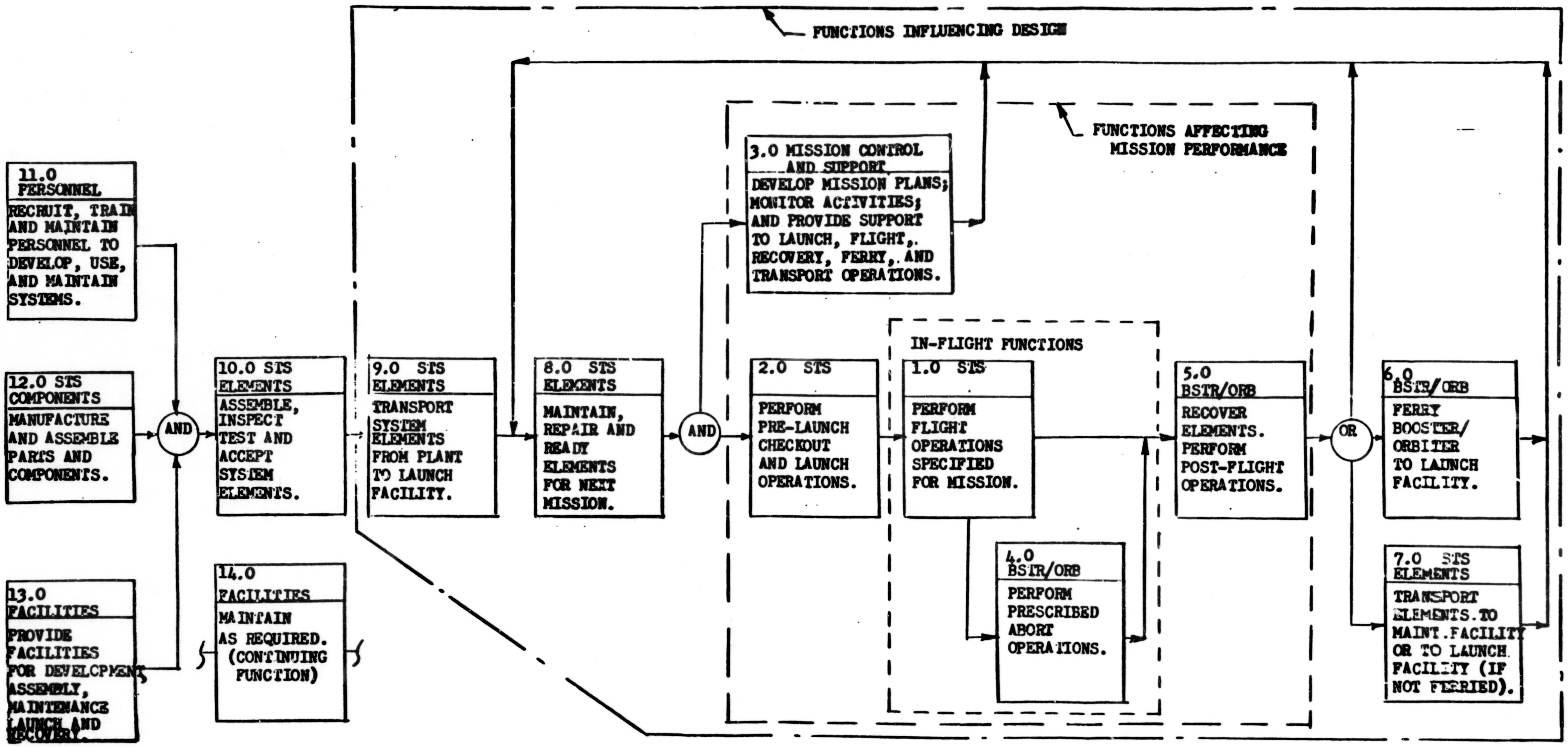

Figure 2.1-1 Top-Level Functions for the Space Transport System Showing Areas Affecting Design, Performance and In-Flight Operations

therefore, there is a feedback loop to FFB 8.0 from the post flight operations of recovery, ferrying and transport as well as from the block designated Mission Control and Support.

Following readying for the mission, the functional flow divides into two branches. One carries the STS through prelaunch and launch operations (FFB 2.0), flight operations (FFB 1.0), and recovery (FFB 5.0). Prelaunch, checkout, and launch operations comprise all those activities required to ready the Shuttle Booster, Shuttle Orbiter, Tug and payload for the mission up to actual liftoff. It includes any changes to the payload, fuel load, or command data required for the vehicle to carry out the mission. Flight Operations (FFB 1.0) includes all those vehicle operations occurring between liftoff and return landing. A no-go functional block (FFB 4.0) is shown leading from flight operations to provide for abort operations. It is recognized that an abort might arise at any point during the flight phase of the mission. Since the Tug cannot return to the earth's surface by itself, the abort procedure, once the Tug is separated from the orbiter, incorporates a functional block requiring it to determine and carry out those operations which will return it to the Shuttle Orbiter, if that is possible. The Tug return to the Shuttle Orbiter is then referenced to the abort procedure of FFB 4.0.

At the top level, recovery operations are included in FFB 5.0, which follows directly after flight operations. Here, the term "recovery" is used only to designate that set of operations involving safing and securing the Shuttle Booster, Shuttle Orbiter, Tug, and any payloads after landing. The term "retrieval" is reserved to designate those activities relating to repossession of a vehicle in space. For all design reference missions, normal flight operations of the Shuttle Orbiter end with landing at Eastern Test Range (ETR) and, thus, recovery of the Shuttle Orbiter, Tug and any payloads is accomplished at the launch facility and no transport or ferry operations (FFB's 6.0 and 7.0) are required. However, the Shuttle Booster may normally land at some point other than the launch site and, under contingency conditions, the Shuttle orbiter also may land at some point other than planned. In those situations, recovery of the STS elements and the payload is followed by the alternatives of direct return to the maintenance/assembly/launch facility (FFB's 6.0 and 7.0).

The branch paralleling flight operations and recovery is Designated Mission Control and Support (FFB 3.0). Even though the Tug is designed for autonomous operation of the Guidance, Navigation, and Control, ground override capability is provided and an overall control and support function will exist. It will include activities such as development of mission plans and schedules, preparation of command data for each mission, establishment of criteria and decision logic for handling mission contingencies, monitoring and recording of actual mission events, performance of launch and recovery operations, and aid to any transport or ferrying activities involving return of STS elements and paylaod to the assembly/launch site for their next mission use.

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2.2 TIMELINES

Tug timelines have been developed for both ground and flight operations. Ground operations timelines include, (1) the operations associated with the initial delivery of the Tug from the manufacturing site, prelaunch, and launch operations, and (2) the turnaround Tug operations, including mission return landing, post-landing safing, unloading, maintenance, and post-maintenance test. Flight operation timeline has been developed for the baseline mission wherein a 3,000 pound payload is delivered to synchronous orbit and a second 3,000 pound payload in synchronous orbit is retrieved and returned to the Shuttle Orbiter payload from synchronous orbit. Projected timelines were also prepared for payload delivery and payload recovery missions.

Function numbers called out on the timelines are those used in the first and second level functional flow block diagrams of Appendix A of this report to provide traceability to the originating requirement. Elapsed time at the end of each operational phase is indicated adjacent to the time bar.

2.2.1 Flight Operations Timelines

Figure 2.2-1 presents a gross timeline history for the baseline mission formulated from flight performance and the functional analysis data. Flight operation is defined to commence with STS liftoff and terminate upon the landing of the Shuttle Orbiter with the Tug/payload. A flight profile for the baseline mission is presented in Figure 2.2-2 with the major mission events identified by event numbers.

The operation times from liftoff to circularization in the 100 N.Mi. parking orbit are assumed to be the same for all missions. This mission phase is depicted by Events 1 through 5 on Figure 2.2-2 and by FFB 1.1 through 1.4.4 on the timeline. Following deployment of the Tug/payload (TUG/PL) from the Shuttle Orbiter cargo bay, the Tug/payload will be checked (Event 6) for mission readiness prior to and subsequent to detachment from the Shuttle Orbiter (FFB 1.4.7 through 1.4.11). The phasing coast operation (cross-hatched) time (Event 7) can vary from zero to 12 hours while waiting for Ascent Ellipse Injection (AEI) burn to place the payload at the desired longitude in synchronous orbit (FFB 1.7). The main engine burn duration for AEI is approximately 1,290 seconds (FFB 1.8.7). The ascent ellipse coast is bounded by events "AEI Burn" (Event 8) and Synchronous Orbit Insertion (SOI) burn (Event 10) with Midcourse Correction (MCC) burn (Event 9) providing for midcourse correction. The midcourse correction burn performed by the Auxiliary Propulsion System (APS) has a duration of approximately 199 seconds. The ascent ellipse times are identified by FFB's 1.8.1 to 1.9.7 in Figure 2.2-1.

A main engine burn of approximately 560 seconds places the Tug/payload into a circular synchronous orbit. One hour is allocated for fine orbit adjust and deployment of the payload (Event 11). An APS burn of approximately 231 seconds initiates the rendezvous (Events 12-15) with the payload in synchronous orbit to be retrieved by the Tug. The phase-in-orbit (Event 12) time during rendezvous can vary from zero to 72 hours depending upon the location of the payload to be retrieved. The time bar for FFB 1.19.8 coast in rendezvous

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Figure 2.2-1 Mission Operations Timeline (Baseline Mission)

FFB NO.'S	OPERATION PHASE	TIME (HOURS)
1.1-1.4.4	Boost to 100 N. Mi. Circular Orbit	0.85
1.4.4-1.4.7	Activate & Check Tug Subsystems	1.05
1.4.7-1.4.8	Extend Deployment Mechanism	1.43
1.4.8-1.4.9	Uncouple Tug and Separate	1.77
1.7	Perform Phasing Coast Operations	14.15
1.8.1	Perform AEI Main Engine Burn	14.51
1.8.1-1.8.7	Coast to MCC Burn	16.42
1.8.7	Perform MCC Burn	16.47
1.8.7-1.9.7	Coast to SOI Burn	19.67
1.9.7	Perform SOI Main Engine Burn	19.82
1.9.7-1.10.5	Orbit Adjust	20.05
1.10.5-1.19.5	Deploy Payload to 2nd PL Retrieval	20.88
1.19.5	2nd PL Retrieval Burn	20.94
1.19.5-1.19.9	Coast in Rendezvous Orbit	94.96
1.19.9	Perform Rendezvous Burn	95.02
1.19.9-1.19.19	Dock and Attach PL	95.52
1.19.19-1.19.20	Safe Payload	96.02
1.11.1-1.11.6	Phase Coast to DEI	108.55
1.11.6	Perform DEI Main Engine Burn	108.64
1.11.6-1.11.8	Coast to MCC Burn	110.79
1.11.8	Perform MCC Burn	110.81
1.11.8-1.12.7	Coast to POI Burn	114.16
1.12.7	Perform POI Main Engine Burn	114.24
1.12.7-1.13.7	Perform Phasing Orbit Coast Operations	136.41
1.13.7	Perform OOTI Main Engine Burn	136.41
1.13.7-1.13.9	Coast to OOI Burn	137.33
1.13.9	Perform OOI Main Engine Burn	137.33
1.13.9-1.14.4	Terminal Phase Initial	138.43
1.14.4-1.14.5	Terminal Phase Final	139.45
1.14.5-1.15.4	Perform Post Docking Orbit Operations	156.01
1.15.4-1.1[illegible]	Deorbit to Landing	156.42

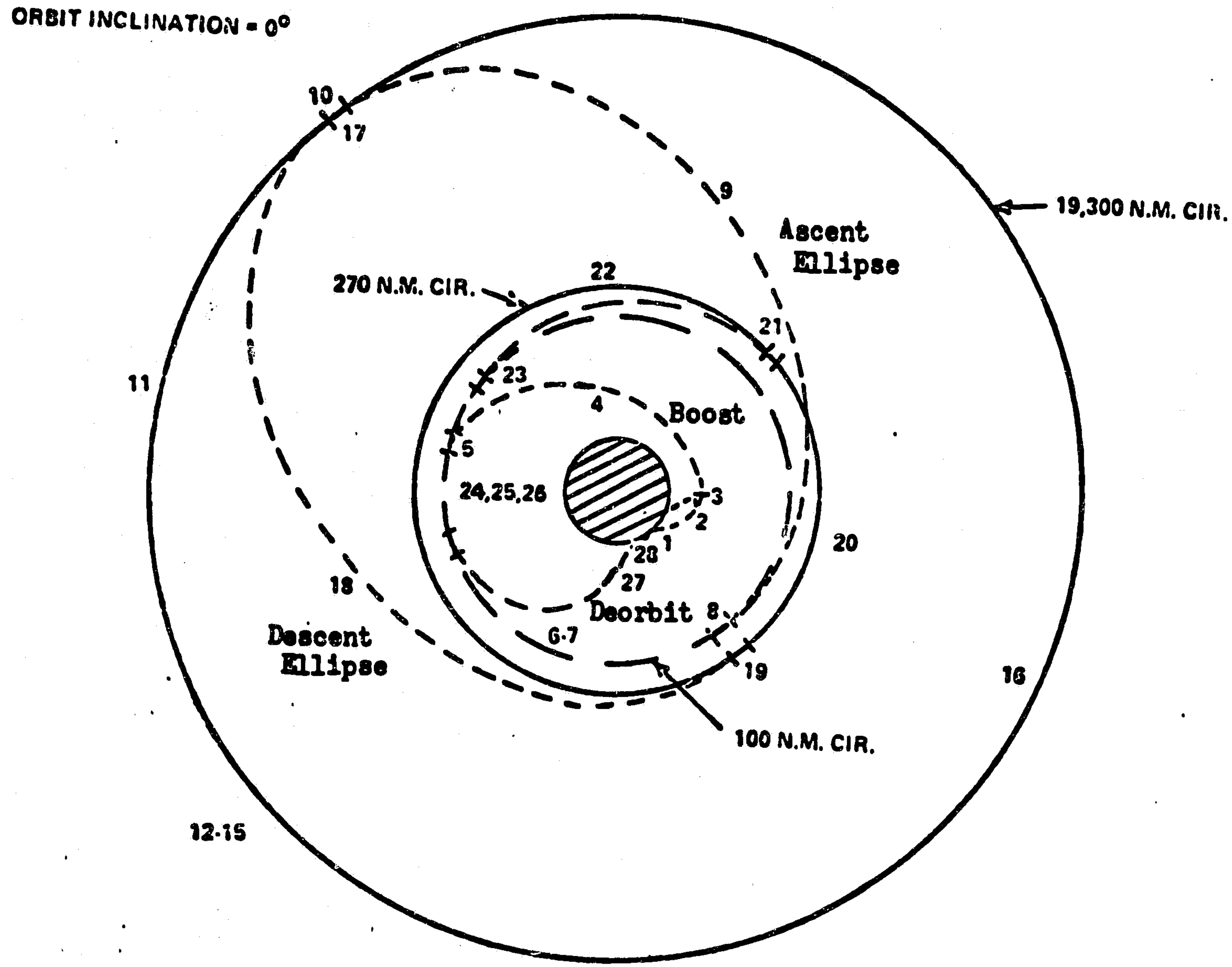

Figure 2.2-2 Baseline Mission Profile and Event Sequence

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includes the maximum phasing coast time (72 hours) and the rendezvous time (2 hours). The second rendezvous burn and the docking burn is accomplished by the APS with burn durations of approximately 227 seconds and 34 seconds respectively. The phase coast operations or FFB's 1.19.19 to 1.11.6 include the half hour allocated for safing the payload (Event 15) and the phasing coast (Event 16) to Descent Ellipse Injection (DEI) burn. As in the case of coast prior to AEI, the coast period prior to DEI burn can vary from zero to 12 hours. The DEI burn is performed by the main engine with a burn duration of approximately 358 seconds.

The Descent Ellipse (18) is bounded by Event 17 (DEI burn) and 19 (Phasing Orbit Insertion-POI) as indicated in Figure 2.2-2. A midcourse correction burn (Event 18) of approximately 87 seconds (FFB 1.11.8) is accomplished during the descent to the 270 N.Mi. phasing orbit. The main engine is utilized for circularization (Event 19) in 270 N.Mi. phasing orbit, Operations Orbit Transfer Injection (OOTI) (Event 21), and Operations Orbit Insertion (OOI) (Event 23) with engine burn duration of 305 seconds, 8.5 seconds and 8.3 seconds respectively.

A total of three hours is allocated for Terminal Phase Initial (TPI), Terminal Phase Final (TPF), and docking operations (Events 25, 25). The in-orbit phasing coast time of the Shuttle Orbiter can vary from zero to 15 hours with the worst case condition of 15 hours shown on the timeline. Forty two minutes (0.7 hr) is allocated for deorbit to landing (Events 26-28) of the Shuttle Orbiter.

The projected timelines of the payload delivery mission and the payload retrieval mission, Figures 2.2-3 and 2.2-4 respectively, are derived from the baseline mission with the following modifications. The 72 hour rendezvous phasing coast time in synchronous orbit is not included in either the payload delivery or payload retrieval missions. Excluded from the payload delivery timeline are the functions of payload rendezvous, dock with payload, and safe payload. In the payload retrieval mission, the time allocation for payload deployment is deleted.

2.2.2 Ground Operations Timelines

Two gross timelines have been prepared for ground operations. The first ground operation timeline is representative of those operations associated with the initial delivery of a Tug at the launch site from the manufacturing site. The second ground operation timeline is representative of the Tug turnaround operations which begins with landing of the Shuttle Orbiter and Tug/payload and terminates with completion of post maintenance testing.

Ground operations time estimates starting with the "Accumulation of Tug Elements" and continuing through "Loading of Tug/PL in the Orbiter Cargo Bay" are derived from the baseline Tug configuration and operations defined in the Tug Ground Operations FFBD. The remaining ground operations up to the time of launch are designed to be compatible with the Shuttle Booster/Orbiter operations following the loading of the Tug/PL in the Shuttle Orbiter cargo bay.

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Figure 2.2-3 Mission Operations Timeline (8.06 K Payload Delivery)

F.F.B. NO.'S	OPERATION PHASE	TIME (HOURS)
1.1-1.4.4	Boost to 100 N.Mi. Circular Orbit	0.85
1.4.4-1.4.7	Activate & Check Tug Subsystem	1.05
1.4.7-1.4.8	Extend Deployment Mechanism	1.43
1.4.8-1.4.9	Uncouple Tug and Separate	1.77
1.7	Perform Phasing Coast Operations	14.15
1.8.1	Perform AEI Main Engine Burn	14.51
1.8.1-1.8.7	Coast to MCC Burn	16.42
1.8.7	Perform MCC Burn	16.48
1.8.7-1.9.7	Coast to SOI Burn	19.48
1.9.7	Perform SOI Main Engine Burn	19.63
1.9.7-1.10.5	Orbit Adjust	20.01
1.10.8	Deploy Payload	20.84
1.10.8-1.11.6	Coast to DEI	33.47
11.6	Perform DEI Main Engine Burn	33.54
1.11.6-1.11.8	Coast to MCC Burn	35.69
1.11.8	Perform MCC Burn	35.71
1.11.8-1.12.7	Coast to POI Burn	39.26
1.12.7	Perform POI Main Engine Burn	39.32
1.12.7-1.13.7	Perform Phasing Orbit Coast Operations	61.49
1.13.7	Perform OOTI Main Engine Burn	61.49
1.13.7-1.13.9	Coast to OOI Burn	62.41
1.13.9	Perform OOI Main Engine Burn	62.41
1.13.9-1.14.4	Terminal Phase Initial	63.51
1.14.4-1.14.5	Terminal Phase Final	64.54
1.14.5-1.15.4	Perform Post Docking Orbit Operations	81.1
1.15.4-1.16	Deorbit to Landing	81.8

Figure 2.2-4 Mission Operations Timeline (4.16 K Payload Retrieval)

F.F.B. NO.'S	OPERATION PHASE	TIME (HOURS)
1.1-1.4.4	Boost to 100 N.Mi. Circular Orbit	0.85
1.4.4-1.4.7	Activate & Check Tug Subsystem	1.05
1.4.7-1.4.8	Extend Deployment Mechanism	1.43
1.4.8-1.4.9	Uncouple Tug and Separate	1.77
1.7	Perform Phasing Coast Operations	14.15
1.8.1	Perform AEI Main Engine Burn	14.49
1.8.1-1.8.7	Coast to MCC Burn	16.40
1.8.7	Perform MCC Burn	16.41
1.8.7-1.9.7	Coast to SOI Burn	19.61
1.9.7	Perform SOI Main Engine Burn	19.76
1.9.7-1.10.5	Orbit Adjust	19.99
1.19.9-1.19.19	Dock and Attach Payload	20.82
1.19.19-1.19.20	Safe Payload	21.32
.11.1-1.11.6	Phase Coast to DEI Burn	33.85
1.11.6	Perform DEI Main Engine Burn	33.96
1.11.6-1.11.8	Coast to MCC Burn	36.11
1.11.8	Perform MCC Burn	36.14
1.11.8-1.12.7	Coast to POI Burn	39.49
1.12.7	Perform POI Main Engine Burn	39.59
1.12.7-1.13.7	Perform Phasing Orbit Coast Operations	61.67
1.13.7	Perform OOTI Main Engine Burn	61.67
1.13.7-1.13.9	Coast to OOI Burn	62.59
1.13.9	Perform OOI Main Engine Burn	62.59
1.13.9-1.14.4	Terminal Phase Initial	63.69
1.14.4-1.14.5	Terminal Phase Final	64.72
1.14.5-1.15.4	Perform Post Docking Orbit Operations	81.28
1.15.4-1.16	Deorbit to Landing	81.98

The first six working days shown on the timeline of Figure 2.2-5 reflects two eight hour working shifts whereas the last three days prior to launch are "around-the-clock" (three shift) operations. The nine day ground operations timeline will be extended if launch date is not critical and an eight hour single shift working day is employed. Conversely, if time to launch date is critical, the timeline can be contracted through employment of an "around-the-clock" three shift working day. Task duration time is placed adjacent to each time bar to facilitate reading the chart.

Tug elements such as aft interstage and Tug with engine support equipment will be accumulated at the launch site and readied for assembly into the Tug flight configuration. These activities include unloading of the elements from the transportation vehicles, removal of environmental protective covers, and visually inspecting the elements for potential shipment or handling damage. Following removal of the engine support equipment and installation of the aft interstage, the Tug subsystems will be functionally checked to verify operational readiness. A 12 hour contingency time is allocated for performing repairs to malfunctioning subsystems (FFB 8.2.3). This is followed by a final test prior to mating the payload to the Tug.

Operations following the initial acceptance of the Tug or post maintenance tests are common to both initial Tug delivery ground operations and turnaround ground operations between Tug missions. A Tug/payload (Tug/PL) integration system test following Tug/PL mating operations is performed to verify physical and functional operational readiness of the Tug/PL interfacing subsystems. The Tug/PL combination is then serviced and transported to the Shuttle Orbiter loading bay for installation in the Orbiter's cargo bay. The capability must also exist for installation and removal of the Tug/PL in the cargo bay with the Shuttle Orbiter in the vertical position. Once installed in the Shuttle Orbiter's cargo bay, the Tug/PL activities are paced to the Shuttle prelaunch operations which include; Tug/PL to Shuttle Orbiter interface verification, Shuttle Booster/Orbiter erection and mating, roll out of the STS to the launch pad, final STS readiness checkout, countdown (including propellant loading as part of total vehicle loading procedure) and launch,

The Tug turnaround activity phasing with projected time goals is presented in Figure 2.2-6. The ground operations, from landing through post-maintenance testing, are included in the turnaround timeline. Post-landing safing and securing phase activities will include inerting and venting propellant tanks and purging of insulation. Purging operations will be continuous during ground operations to prevent degradation of insulating material.

The Tug and payload will be transported to the Shuttle Orbiter maintenance area where they will be removed and loaded on a transporter. After transporting the Tug and payload to the Tug maintenance area, they will be separated. The payload will be transported to its processing area and the Tug placed on a maintenance stand. (The maintenance stand and transporter may be one and the same.

Turnaround maintenance will be accomplished in 54 working hours from the time the Tug is delivered to the maintenance facility. This time is considered to be the maximum required during the operational phase. There will be many

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Figure 2.2-5 Ground Operations Timeline (Initial Tug Launch After Delivery From Manufacturing)

FFB NO.	FUNCTION	WORKING DAYS (2 SHIFTS) (3 SHIFTS) 1–10
8.1.1	Accumulate Tug Elements	
8.1.1.1	Unload Tug Interstage and Move to Assy. Area	4
8.1.1.2	Unload Tug and Move to Assy. Area	4
8.1.1.3	Remove Tug Environmental Covers	2
8.1.1.4	Remove Engine Protective Frame	3
8.1.1.5	Visually Inspect Tug and Interstage	3
8.1.2	Mate Elements Into Tug	6
8.2.1	Mount Tug on Diagnostic/Maintenance Stand	
8.2.1.1	Position Tug on Stand, Position Work and Access Platforms	8
8.2.1.2	Make GSE Connections	2
8.2.2	Perform Diagnostic Checks and Determine Maint. Required	4
8.2.3	Perform Required Maintenance	12
8.2.4	Perform Required Testing (Acceptance)	
8.2.4.1	Perform Leak Checks	8
8.2.4.2	Perform Simulated Flight Checks	6
8.3.1	Perform Tug/PL Mating Operations	4
8.3.2	Perform Tug/PL Integration Sys. Test	2
8.3.3	Service Tug/PL (Also install Tug Heat Shield)	8
8.3.4	Transport Tug/PL to Orbiter Loading Bay	2
REF	Orbiter Premate Check	8
8.3.5	Load Tug/PL Into Orbiter	4
8.3.6	Perform Tug/PL to Orbiter Interface Verification	2
8.4	Transport to Launcher and Erect	
REF	Tow Booster to Mating Bay	2
REF	Tow Orbiter to Mating Bay	2
REF	Erect and Install Booster and Orbiter on LUT	
2.1.1	Perform Post Erect Tug/PL STS Checkout	2
2.1.2	Rollout to Launch Pad	6
REF	Mate LUT/Pad Interfaces	8
REF	Verify LUT/Pad Interfaces	4
2.1.3	Perform Prelaunch Readiness Checkout	6
REF	Tank Purge and Ordnance Hook-Up	6
2.2	Countdown and Launch	5

FFD NO.	DESCRIPTION OF FUNCTION	WORK DAYS (TWO SHIFTS) 1 2 3 4 5 6 7
5.1	Perform Post-Landing Safing and Securing	
5.1.1	Safe Tug/Payload	4
5.1.2	Inert Tanks	6
5.1.3	Purge Insulation	Continuous
5.2	Unload Tug/Payload From Orbiter	8
5.2.1	Unload Payload From Tug	4
7.1	Prepare Tug for Transportation to Maint. Area	2
7.2	Load and Transport Tug	2
7.3	Unload Tug	2
8.2	Perform Checkout and Maintenance	
8.2.1	Mount Tug on Maintenance Stand	6
8.2.1.1	Position Work Platforms, Ladders, Etc.	4
8.2.1.2	Connect GSE	2
8.2.3	Perform Preventive Maintenance	
8.2.3.1	Run Subsystem Functional Checkout	4
8.2.3.2	Inspect Accessible Areas	4 8
8.2.3.2	Analyze Flight Data	6
8.2.3.4	Perform Scheduled Replacement, Adjustment, Service	16 16
8.2.3.5	Perform Corrective Maintenance	8 16
8.2.4	Perform Required Testing	8
8.3 (Ref)	Accumulate STS Elements	REF.

Figure 2.2-6 Ground Operations Timeline Landing thru Maintenance

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turnaround cycles accomplished in less time due to proper subsystem operation. The time is, however, not beyond that required early in the operational program when all subsystems will be subjected to close secrutiny. As confidence in the subsystem performance is increased, the amount of inspection and checking will be decreased.

In the maintenance facility, the vehicle will be subjected to diagnostic tests to establish subsystem performance status, visual and nondestructive inspection will be performed, and the flight data will be reviewed. From these functions, the required corrective maintenance actions will be identified to restore the subsystem to an operating condition. As the corrective maintenance requirements are identified, they will be scheduled for accomplishment in parallel with other preventative maintenance activities to the maximum extent practicable.

Items that are found to be discrepant during the above checks and require disposition or repair will be removed and transferred to the supporting shops for failure analysis and disposition. This Level II maintenance provides for the removal, replacement, repair, calibration, adjustment, checkout, test and inspection to the subassembly or part level. If major repair or overhaul is required beyond the capabilities of the second level, the Tug will be transferred to a Level III activity for disposition.

Following the maintenance activities, the vehicle will be subjected to post-maintenance tests that will verify the functional integrity of all subsystems. As experience in the operational program increases, confidence in the subsystem performance will permit tests of only those subsystems disturbed during maintenance.

2.3 MISSION EXPENDABLES SCHEDULE

Table 2.3-1 presents a summary of consumables expended in executing the baseline mission which was derived from the detailed expendable schedule of Appendix D. The history of consumables expended begins with liftoff and ends with touchdown at the landing site. The first three columns identify the mission maneuver phases, the corresponding function flow diagram block numbers and the duration of each maneuver phase. The next five columns identify the Tug initial weight prior to the translation maneuver, the delta-V associated in the maneuver, impulse propellant expended (MPS chilldown propellant included), the propulsion mode utilized to execute the maneuver, i.e., Main Propulsion System (MPS) or Reaction Control System (RCS) (RCS is used interchangably with APS - Auxiliary Propulsion System) and the burn duration in seconds of each translation maneuver. A two percent contingency delta-V of 572 feet per second was proportionally distributed among the MPS burns required for injection and circularization thrusting maneuvers in excess of 100 feet per second.

Maneuvers performed by RCS (APS) other than orbit translation maneuvers are Attitude Hold, Attitude Maneuver, and Propellant Settling. These maneuver functions are interspersed throughout the entire mission as required by mission operations and are summed as a single line entry for each mission maneuver phase in Table 2.3-1. A propellant settling maneuver is executed prior to each main

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Table 2.3-1 Tug Expendables Schedule Summary (Baseline Mission)

Mission Maneuver Phase	F. F. B No. 's	Phase Duration (Hrs)	Weight After Mission Phase (Lbs)	Delta-V Trans-lation (Fps)	Impulse Propellant (Lbs)	Prop-ulsion Mode	Burn Time (sec)	Other RCS (Lbs)	Fuel Cell & Cond. (Lbs)	Pay-load (Lbs)	Major Mission Phases
Liftoff thru 100 N. Mi. Circular Orbit	1.1-1.4.4	0.85	65000.	-	-	-	-	-	-	3000	Boost to Orbit
Circuliarization thru Tug-Shuttle Separation	1.4.4-1.4.9	0.92	① 63544.3	10	52.4	RCS	71	1.1	1.2	3000	Orbital
Separation thru AEI Burn	1.7.1-1.8.1	12.38	35926.8	8615	27,577.3	MPS	1290	23.3	16.9	3000	Operations
AEI Burn thru MCC	1.8.1-1.8.7	2.30	35773.3	④ 50	146.5	RCS	199	4.2	2.8	3000	Transfer to
MCC thru SOI	1.8.7-1.9.7	3.20	24030.5	6010	11,722.8	MPS	552	15.5	4.5	3000	Sync. Orbit
SOI thru Orbit Adjust	1.9.7-1.10.5	0.38	23971.2	30	59.3	RCS	80	0.2	0.2	3000	
Orbit Adjust thru PL Deploy	1.10.5-1.10.8	0.14	20951.0	10	17.2	RCS	23	2.7	0.2	0	
PL Deploy thru 2nd PL Retrieval Burn	1.10.8-1.19.7	0.67	20778.4	④ 100	170.4	RCS	231	1.3	0.9	0	Sync Orbit
2nd PL Retrieval Burn thru PL Rendez	1.19.7-1.19.9	74.08	20463.4	④ 100	167.8	RCS	228	48.7	99.5	0	Operations
PL Rendez. thru PL Dock	1.19.9-1.20.16	0.56	23435.1	15	25.2	RCS	34	2.2	0.9	3000	
PL Dock thru DEI	1.20.16-1.11.6	13.02	② 15749.2	5937	7,569.3	MPS	356	16.2	17.4	3000	
DEI thru MCC	1.11.6-1.11.8	2.25	15678.3	50	64.2	RCS	87	3.7	3.0	3000	Transfer to
MCC thru POI	1.11.8-1.12.7	3.37	9213.9	8025	6,448.3	MPS	303	11.6	4.5	3000	Phasing Orbit
POI thru OOTI	1.12.7-1.13.7	22.24	8990.3	302	179.3	MPS	8	14.5	29.8	3000	Transfer to
OOTI thru OOI	1.13.7-1.13.9	0.92	8805.0	302	175.7	MPS	8	8.3	1.3	3000	Oper. Orbit
OOI thru TPI	1.13.9-1.14.4	1.10	8764.9	④ 50	35.9	RCS	49	3.1	1.4	3000	
TPI thru TPF and Dock	1.14.4-1.14.5	1.01	8727.5	④ 50	35.6	RCS	48	0.3	1.4	3000	Oper. Orbit
Dock thru Deorbit	1.14.5-1.16	16.56	③ 8328.0	-	-	-	-	-	-	3000	Dock & Safe
Deorbit thru Landing	1.16-5.0	0.70	8328.0	-	-	-	-	-	-	3000	Deorbit & Land

① 1400 Pounds Remain With Shuttle Orbiter

② 83 Pounds of Propellant Vent (Thermodynamic)

③ Dump Propellant and Safe Tanks

④ Multiple Burns

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engine burn and propellant expended for this function is included in the "Other RCS (APS) Maneuver Column."

Tug hydrogen and oxygen are utilized by the fuel cell for generating electricity and the propellant feedlines are chilled by a thermodynamic vent conditioning system by expending hydrogen. The fuel cell consumes 0.824 pounds of reactants per hour with gas generator propellants for APS accumulator recharge of 0.12 pounds per hour and feed line conditioning of approximately 0.4 pounds of expendables per hour. These expendables rates are combined and shown in the fuel cell and conditioning column. The next to the last column of Table 2.3-1 identifies the time the Tug has a payload attached and the last column identifies major mission phases.

Heat input to the Tug propellant from liftoff to the time of main engine burn for descent ellipse injection (DEI), is dissipated by a venting of propellant gases after propellant settling just prior to the main engine DEI burn. A total of 83 pounds of propellant gases (48 pounds of GH_2 and 55 pounds of GOX) are vented and is reflected in the "initial weight" column in line with "PL Dock to DEI" (FFB No.'s 1.20.16-1.11.6). See Note 2 on Table 2.3-1.

Approximately 1400 pounds of Shuttle Orbiter-to-Tug Docking structure and safing equipment/gases remain in the Shuttle Orbiter when the Tug is deployed. Upon Tug return and docking with the Shuttle Orbiter, the remaining propellants are dumped and the tanks "safed" through a tank pressurization, blowdown and repressurization cycle. The final weight, 8328 pounds, returned to earth by the Shuttle Orbiter includes Tug with residual (tank pressurization) gases and retrieved payload.

The detailed expendables schedule will be found in Appendix D.

3.0 FLIGHT PERFORMANCE

The baseline mission requires that the Tug, which has a burnout weight of less than 6000 pounds, deliver a 3000 pound payload to geosynchronous orbit and retrieve a similar payload returning it to the original 100 n.mi. orbit. The demanding nature of this mission dictated that an exceptionally high mass fraction would have to be attained in the design of the Tug. Consequently, a performance study plan was established which not only included point design precision performance simulations but also investigated a number of performance enhancement techniques.

The study approach used is presented in Figure 3.0-1. Since precision performance simulations were of primary importance to the point design analysis, it was important that a set of performance ground rules be established and that the ΔV budget furnished for the study be verified. Next, since the weight of the Tug was of critical importance to the analysis, a general performance chart was established linking the Tug burnout weight to other performance parameters including such factors as propellant boiloff and engine start/stop losses.

In addition to establishing point design performance capability, a number of performance tradeoffs were made to aid in making point design decisions such as selecting optimum insulation thickness.

Finally, a number of performance enhancement techniques were investigated with the intention of providing a further increase in the performance capability of the Tug. It should be noted that, in general, the performance enhancement techniques involved deviation from the established performance groundrules. Altogether, 16 performance enhancement ideas were at least briefly investigated. The four shown in Figure 3.0-1 represent some of the more promising techniques considered.

3.1 POINT DESIGN

3.1.1 Performance Groundrules

Study groundrules specifically affecting the Tug performance are as follows:

1. Shuttle payload capability is 65000 lb

2. Tug is ground based

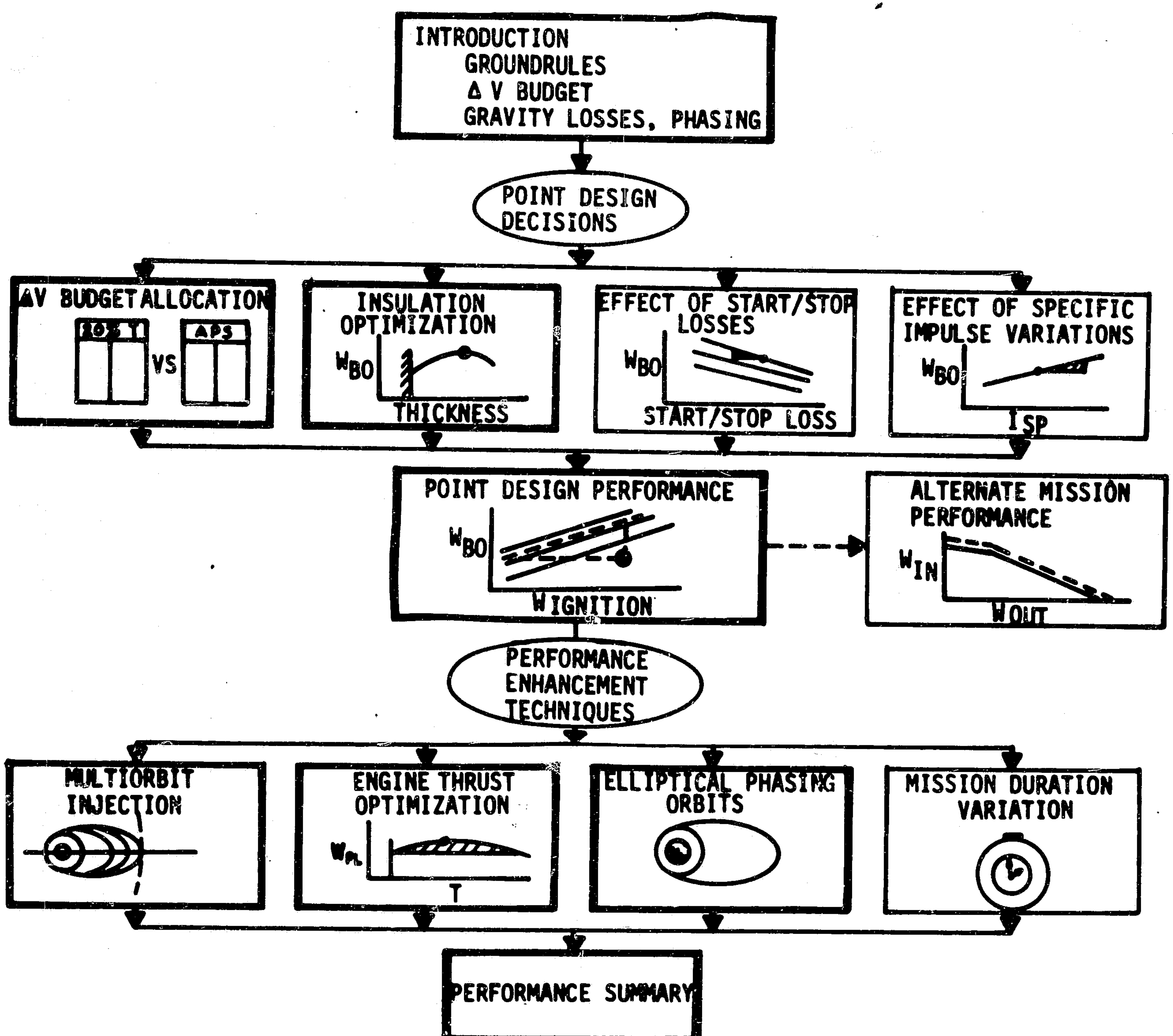

Figure 3.0-1 Tug Flight Performance Studies

3. Mission time in orbit:

   6 days unattached to Shuttle

   1 day standby in Shuttle cargo bay

4. Operating orbit: 100 n.mi. circular, 28.5° inclination

5. Engine characteristics

Engine	Thrust (lb)	Specific Impulse (sec)	Propellants
Main engine	10,000	470	LOX/LH_2
Main Engine Throttled to 20%	2,000	461	LOX/LH_2
RCS	TBD	380	GOX/GH_2

6. Missions Assigned:

Mission	Outbound Payload (lb)	Inbound Payload (lb)
Baseline	3,000	3,000
Alternate Mission No. 1	8,060	0
Alternate Mission No. 2	0	4,160

7. Delta V budget: (See Table 3.1-1 for detailed breakdown)

   Main engine - 29,191 FPS
   Orbit maneuvers - 300 FPS
   RCS translation and attitude control maneuvers - 190 FPS

The ΔV Budget

A breakdown of the NASA baseline mission ΔV budget is shown in Table 3.1-1. For comparison purposes the equivalent ΔV increments as determined by NR are also listed. The deviation between corresponding NASA and NR ΔV entries is in no case more than 10 ft/sec. In most cases it is much less. It should be noted that the three 100 ft/sec ΔV's used to achieve

TABLE 3.1-1. ΔV BUDGET

EVENT	Δ VELOCITY - FT/SEC: MAIN ENGINE NASA	MAIN ENGINE NR	20% THROTTLED* NASA	20% THROTTLED* NR	RCS NASA	RCS NR
SEPARATE FROM SHUTTLE AT 100 N.MI.					10	10
PERIGEE BURN	8,136	8,130				
GRAVITY LOSS	310	310				
MID-COURSE CORRECTION					50	50
APOGEE BURN	5,883	5,887				
GRAVITY LOSS	10	10				
STATIONKEEPING					30	30
DEPLOY PAYLOAD					10	10
INJECT INTO PHASING ORBIT TO RETRIEVE PAYLOAD			100			100**
RETRIEVE PAYLOAD			100		15	115**
DE-ORBIT	5,814	5,818				
GRAVITY LOSS	7	7				
MID-COURSE CORRECTION					50	50
CIRCULARIZE IN 270 N.MI.	7,842	7,836				
GRAVITY LOSS	25	25				
TRANSFER TO SHUTTLE ORBIT 100 N.MI.	592	592				
TERMINAL RENDEZVOUS			100		15	115**
DOCK WITH SHUTTLE AT 100 N.MI.					10	10
CONTINGENCY (2 PERCENT)	572	572				
TOTAL	29,191	29,187	300	0	190	490

*MAIN ENGINE THROTTLED TO 20 PERCENT
**MORE THAN ONE BURN

phasing, retrieval and rendezvous have been transferred from the idle mode column to the RCS column. This change was made because the phasing, retrieval, and rendezvous maneuvers will actually be broken up into two or more smaller burns. The RCS engines are best used for these burns because they provide more realistic burning intervals and negligible start/stop losses.

The ΔV budgets for the two alternate missions are not tabulated in Table 3.1-1. These budgets differ slightly from the baseline mission budget because only a single payload is handled and therefore the ΔV increments allocated for the phasing and rendezvous maneuvers are smaller. For this reason, the total ΔV budgets for the alternate missions are 215 ft/sec lower than the total shown in Table 3.1-1.

Gravity Loss Comparisons

In boosting toward the geosynchronous altitude, the Tug sustains a gravity loss of 310 ft/sec as indicated in Figure 3.1-1. The magnitude of the gravity loss is primarily a function of the velocity increment generated and the initial thrust-to-weight ratio of the Tug. A comparison between the NASA g-loss curve and a comparable set of NR values is shown in Figure 3.1-1. Note that the maximum deviation is less than 10 ft/sec and in the region of interest (between $F/W_o = 1.5$ and $F/W_o = 2.5$) the two sets of values are in even closer agreement. The NR values were obtained by using numerically integrated trajectory simulations.

Tug/Payload Phasing and Rendezvous

Once the Tug has delivered its outbound payload to the geosynchronous orbit, it must retrieve a second payload and carry it back to the 100 n.mi. orbit for rendezvous with the Shuttle. The second payload which is also in geosynchronous orbit, may be as much as 6000 n.mi. from the Tug. At the geosynchronous altitude this is equivalent to a geocentric arc of 15 degrees. In order to achieve rendezvous with the second payload, the Tug must perform a sequence of at least four powered maneuvers; two to achieve phasing and two to achieve rendezvous. The first phasing maneuver consists of a retrograde burn which drives the perigee of the Tug's orbit downward thus shortening its orbital period. Then, once the Tug has coasted into the vicinity of the payload, a posigrade burn at apogee freezes its motion with respect to the payload.

The Tug's phasing rate can be varied by adjusting the distance its perigee is depressed. A greater perigee depression increases the phasing rate. However, this requires a larger total increment (see Figure 3.1-2). The entire 15 degrees of phasing could be made up in one Tug orbit (approximately one day) but the total velocity increment would amount to about 280 ft/sec. Since this exceeds the maximum total 200 ft/sec allotment for the phasing/rendezvous maneuver, phasing cannot be accomplished in a single orbit. However, as illustrated in Figure 3.1-2, phasing can be accomplished in either two or three Tug orbits. A two day phasing interval would require a total ΔV of 140 ft/sec, whereas a three day phasing interval (the maximum

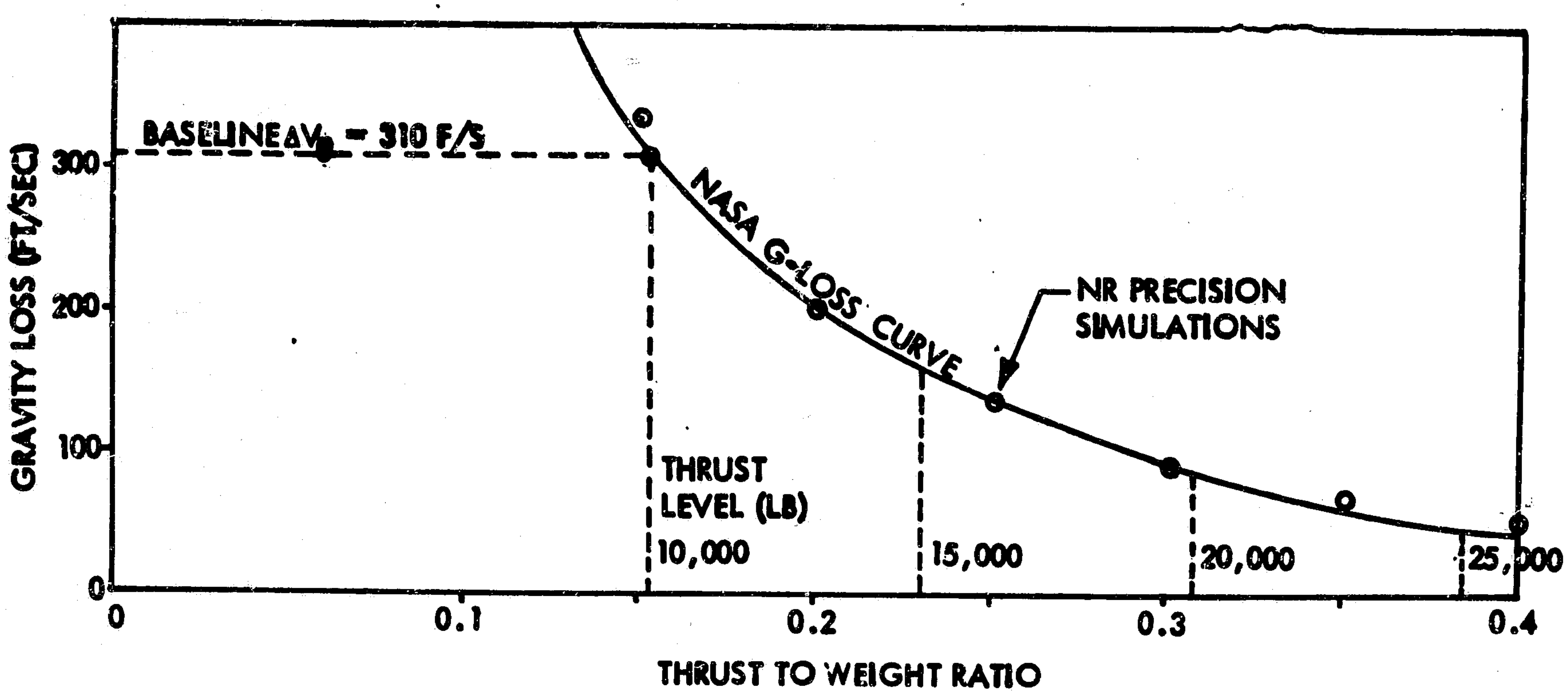

Figure 3.1-1 Tug Gravity Loss Comparisons

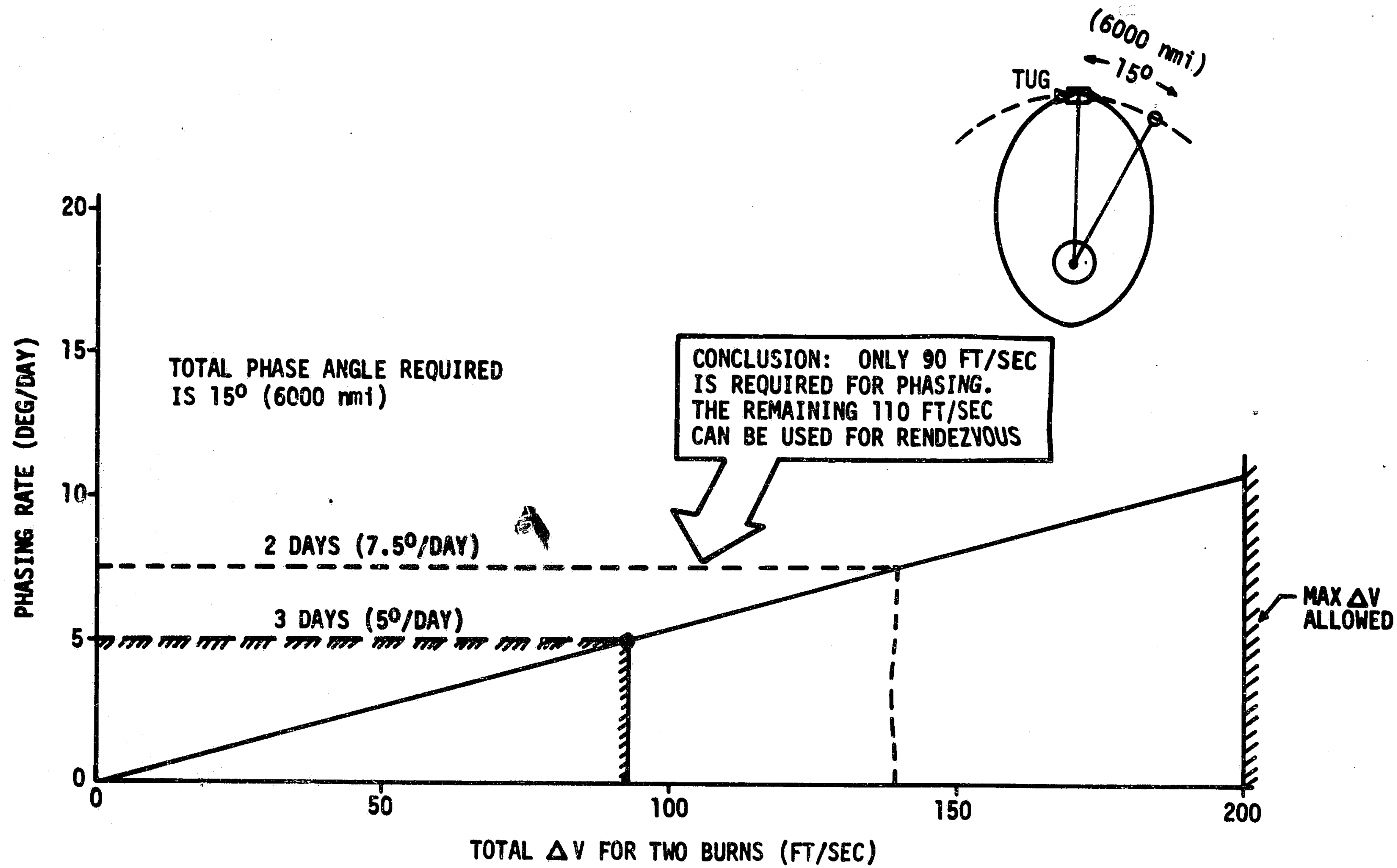

Figure 3.1-2 Tug/Payload Retrieval Phasing Characteristics

allotted in the mission time line) would require about 95 ft/sec. Based on previous study results, the remaining ΔV of 60 to 105 ft/sec appears to be more than adequate for achieving rendezvous within a reasonable time frame.

3.1.2 Baseline and Alternate Mission Performance

The baseline Tug mission calls for a vehicle that can carry a 3000 lb payload into a geosynchronous orbit and return to the 100 n.mi. orbit with a second 3000 lb payload. The payload weights for the alternate missions, which are also flown into geosynchronous orbits, are 8060 lb and 4160 lb. The first alternate mission places the 8060 lb payload in geosynchronous orbit but has no inbound payload. The second alternate mission, by contrast, has no outbound payload but retrieves the 4160 lb payload in geosynchronous orbit and returns it to the Shuttle which is waiting in the 100 n.mi. orbit.

Tug Baseline Performance Capability

Since the Tug design is affected by so many interacting performance parameters, it was important to summarize the design status in a reasonably simplified manner. For example, a change in the high performance insulation system could alter the usable propellant weight, the propellant boiloff rate, and the Tug burnout weight—all of which directly affect the overall performance capability of the Tug. It was found that the effect of all of the important performance parameters could be accounted for on a single performance chart (Figure 3.1-3).

The horizontal axis represents the ignition weight of the Tug and the vertical axis represents the maximum allowable burnout weight. Each of the diagonal lines on the graph represents a specific budget of non-propulsive consumables (i.e., on board fluids which do not provide propulsive energy). The non-propulsive consumables include propellant boiloff losses, start/stop losses, fuel cell reactants, and attitude control propellants. The horizontal scale with the origin in the lower right hand corner of Figure 3.1-3 is used to account for the Tug/Shuttle interface weight. Any weights (such as propellant boiloff weight) that are removed from the Tug prior to Tug ignition would also be accounted for on the lower horizontal scale.

Current values of the actual ignition and burnout weights for the Tug are spotted on the curve together with the current weight of the non-propulsive consumables for the Tug. Measuring vertically from the design point to the allowable burnout weight curve (dashed line) it is seen that 720 lbs of additional burnout weight could be tolerated. In other words, if the Tug burnout weight should increase by 720 lbs, the vehicle would still be capable of flying the baseline mission. On the other hand, if the burnout weight could be held to its current value, then the Tug would be able to carry a payload 720 lb heavier than the baseline payload, i.e., a total payload of 3720 lb.

Measuring horizontally from the design point to the allowable burnout weight curve, it is seen that the ignition weight of the Tug changes by

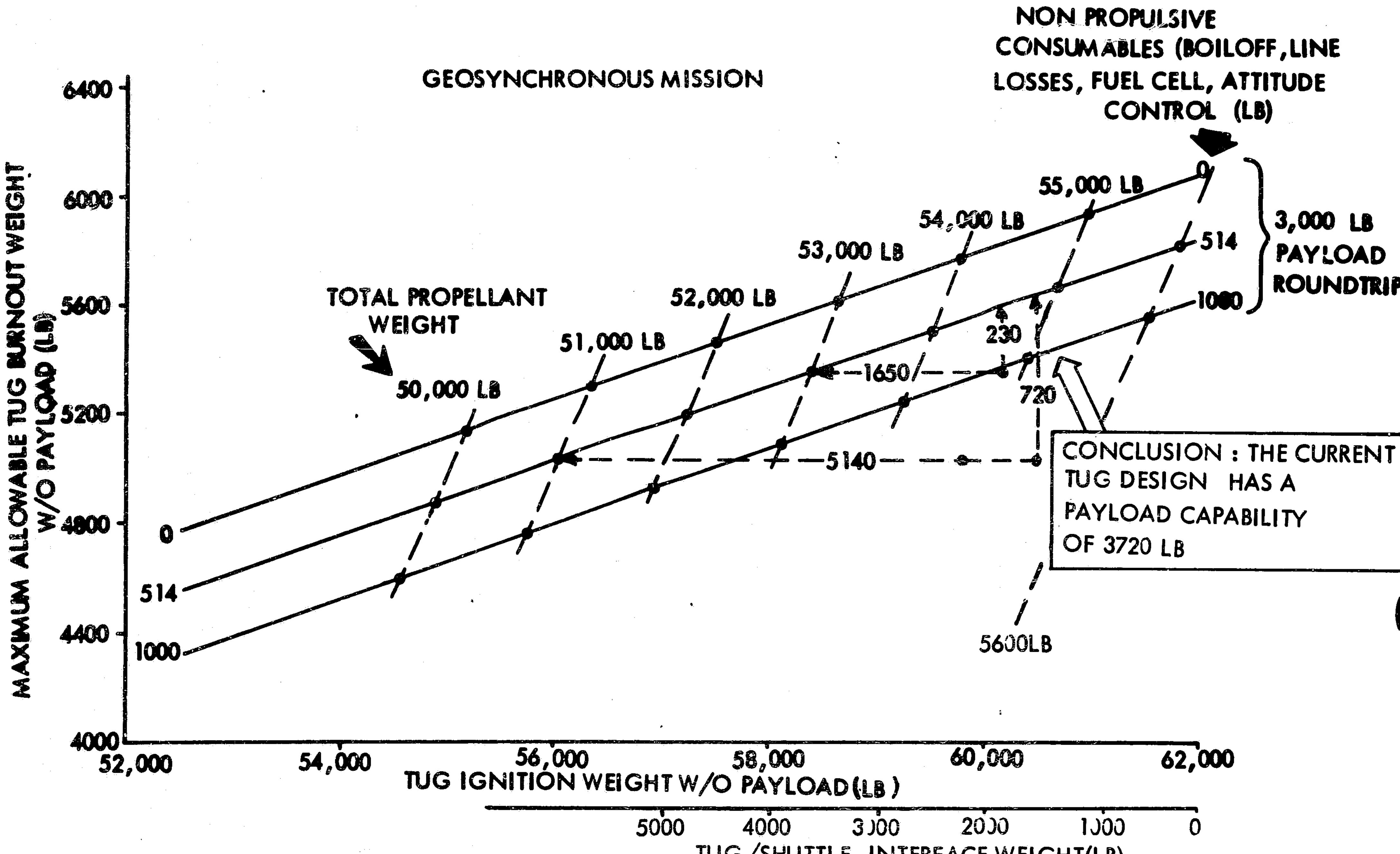

Figure 3.1-3 Tug Baseline Payload Performance Summary

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5140 lb. This can be interpreted in two ways also. It can be visualized as a permissible growth of 5140 pounds in the Tug/Shuttle interface weight or as an allowable reduction of 5140 lb in the Shuttle payload capability.

Alternate Mission Performance Capability

Given a Tug that can fly the baseline mission successfully, the question arises as to whether the same vehicle could also fly the alternate missions. The answer to this question can be obtained from Figure 3.1-4. The solid line represents the inbound/outbound payload trades for missions having the same ΔV budget as the baseline mission. Note that propellant offloading is required when the outbound payload weight exceeds 3000 lb. Off-loading is necessary in order to keep within the 65000 lb payload limit of the Shuttle.

If the alternate missions required the same ΔV budget as the baseline mission, the Tug would be unable to perform the alternate missions since they both fall above the solid diagonal line in Figure 3.1-4. However, since the ΔV budgets for the alternate missions are 215 ft/sec lower than the baseline ΔV budget, the payload trades are as shown by the dashed line and the same vehicle will in fact be able to fly all three missions successfully.

3.1.3 Point Design Performance Tradeoffs

In keeping with the point design aspect of the study, the groundrules specifically tied down most of the critical performance parameters of the Tug. However, there are a few important performance trades that can be carried out within the scope of the study.

Insulation Thickness Tradeoff

During a typical six day mission, approximately 160 pounds of Tug propellant boils off due to heat inputs from various sources. The boiloff propellants represent potentially usable propellant which is lost overboard and can, therefore, no longer be burned. Consequently, a reduction in boiloff rates would increase the amount of available propellant and hence increase the performance capability of the vehicle. However, decreasing the boiloff rate requires that more insulation be added to the vehicle. This in turn, adds weight to the vehicle which reduces the performance capability. This indicates that there might be an optimum insulation thickness representing the best compromise between boiloff loss and insulation weight. Figure 3.1-5 shows the results of a performance tradeoff between boiloff loss and insulation thickness. The minimum insulation thickness of 0.3 inches is dictated by the allowable ground hold propellant boiloff rate.

Three different insulation types considered in the performance analysis are: 1) single aluminized mylar, 2) singly aluminized kapton, and 3) singly goldized kapton. As shown in the figure, the performance gain for optimization of the thickness of any of the insulation types amounts to no more than 20 pounds of allowable burnout weight. Selection of the insulation is discussed in detail in Section 4.2.1 of the report.

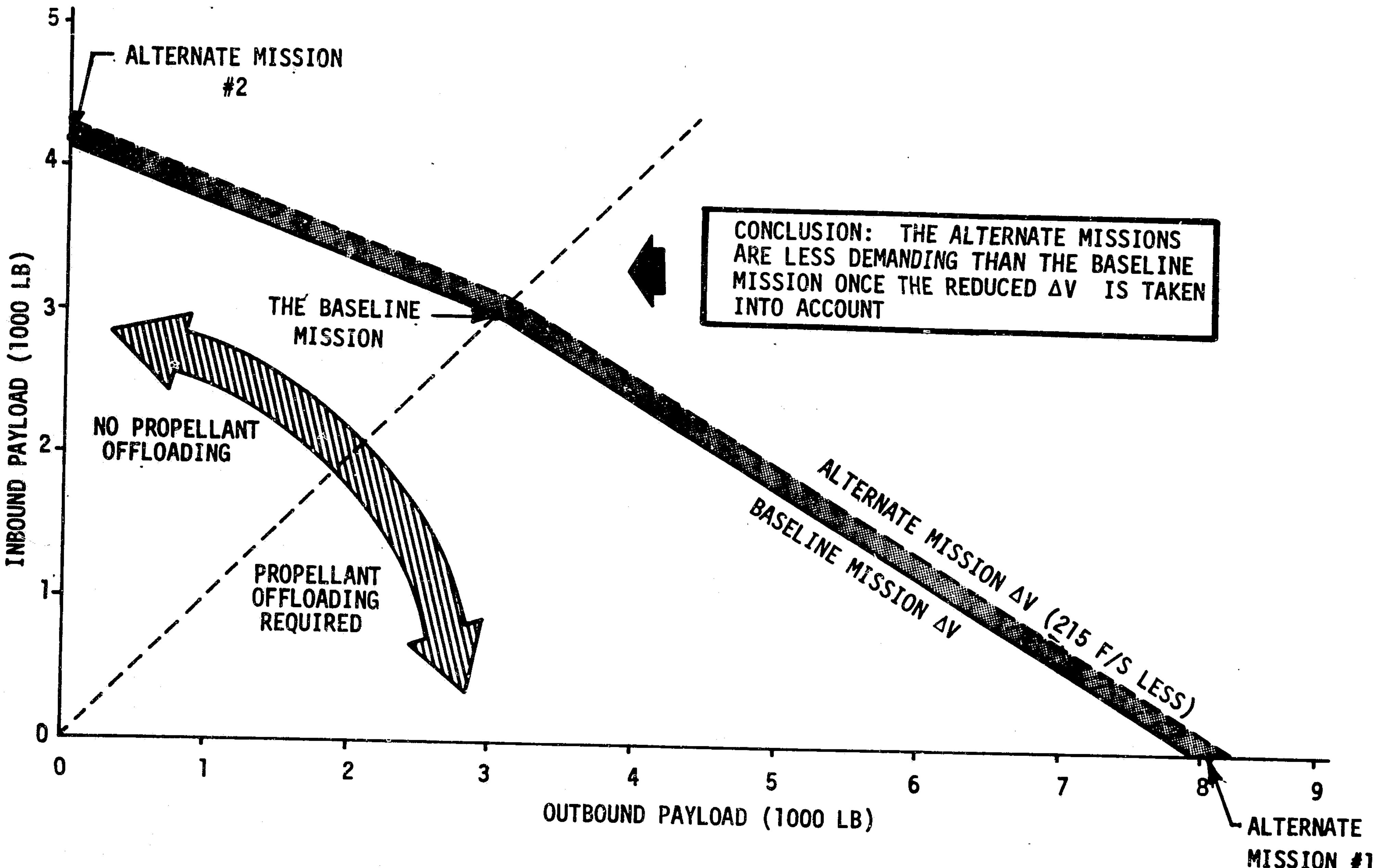


Figure 3.1-4 Inbound/Outbound Payloads Trades

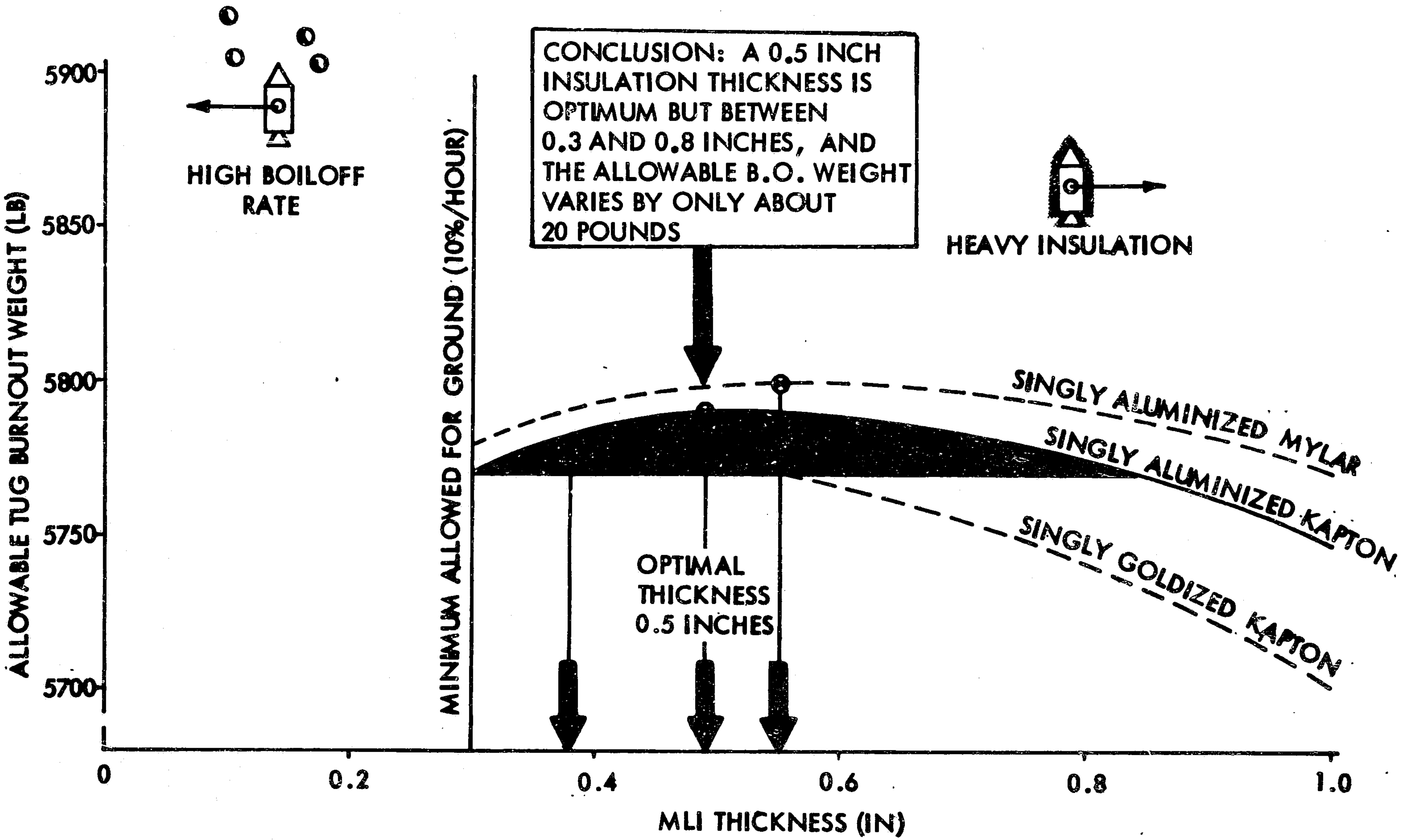


Figure 3.1-5 Insulation Weight Optimization

Effect of Start/Stop Losses

For a typical Tug flight, approximately 85 pounds of propellants are lost due to main engine starts and stops. These losses result, in part, from the fact that at cutoff small quantities of propellants remain trapped and evaporate. Another part of the loss results from operation at reduced thrust during engine chill and from inefficient operation during thrust buildup and decay. For the performance analysis it was assumed that on the average the start/stop losses were approximately 15 pounds per burn. As shown in Figure 3.1-6, these losses result in a reduction of the allowable burnout weight of approximately 75 pounds. Reducing the start/stop losses from 15 pounds per burn to 10 pounds per burn results in an increase of 25 pounds in the Tug allowable burnout weight.

Effect of Specific Impulse Variations

For the point design study, the specific impulse of the Tug main engine was assumed to be 470 seconds. This represents the minimum guaranteed value quoted for the baseline engine. Consequently, the question arises as to what the effect on performance would be if the nominal specific impulse of 473.8 seconds were used instead.

Performance gains that can be achieved by increasing the specific impulse are shown in Figure 3.1-7. An increase of one second in the specific impulse results in a performance gain of 37 pounds of allowable Tug burnout weight. Therefore, if the nominal specific impulse is used rather than the minimum guaranteed, the allowable Tug burnout weight is increased by 141 pounds.

Effect of ΔV Budget Allocation

As was previously stated, the ΔV budget alotted for orbit maneuvers (phasing, retrieval and rendezvous) which was assigned to the main engine throttled to 20 percent in the NASA ΔV budget, was transferred to the RCS system in the NR ΔV budget. The performance gain associated with this change is negligible. However, using the RCS system for the smaller ΔV's required for these maneuvers results in more realistic burning intervals.

Performance Partials

Performance partials or sensitivities were developed for the Tug so that quick evaluations could be made of the effect of various design changes on the performance of the vehicle. The sensitivity of Tug burnout weight to changes in various parameters is presented in Table 3.1-2. It should be noted that Tug burnout weight is interchangeable with payload weight for the .seline 3000 lb round trip payload case. These partials have general applicability to a wide variety of design problems but should be used with some caution. Since the initial weight of the Tug/payload combination is constrained to a maximum value (65000 lb), some of the partials may be interdependent in various subtle ways.

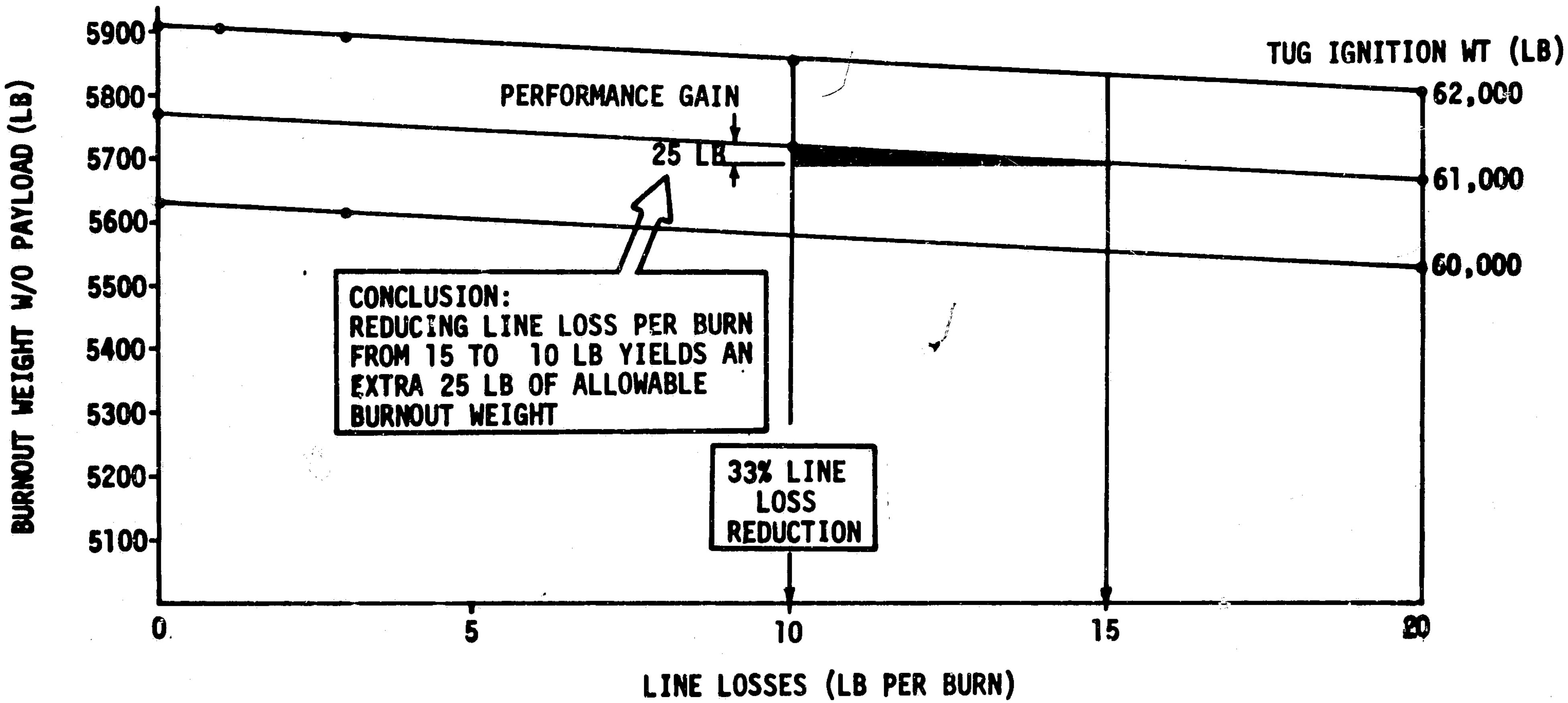


Figure 3.1-6 Effect Start/Stop Losses on Tug Performance

Space Division
North American Rockwell

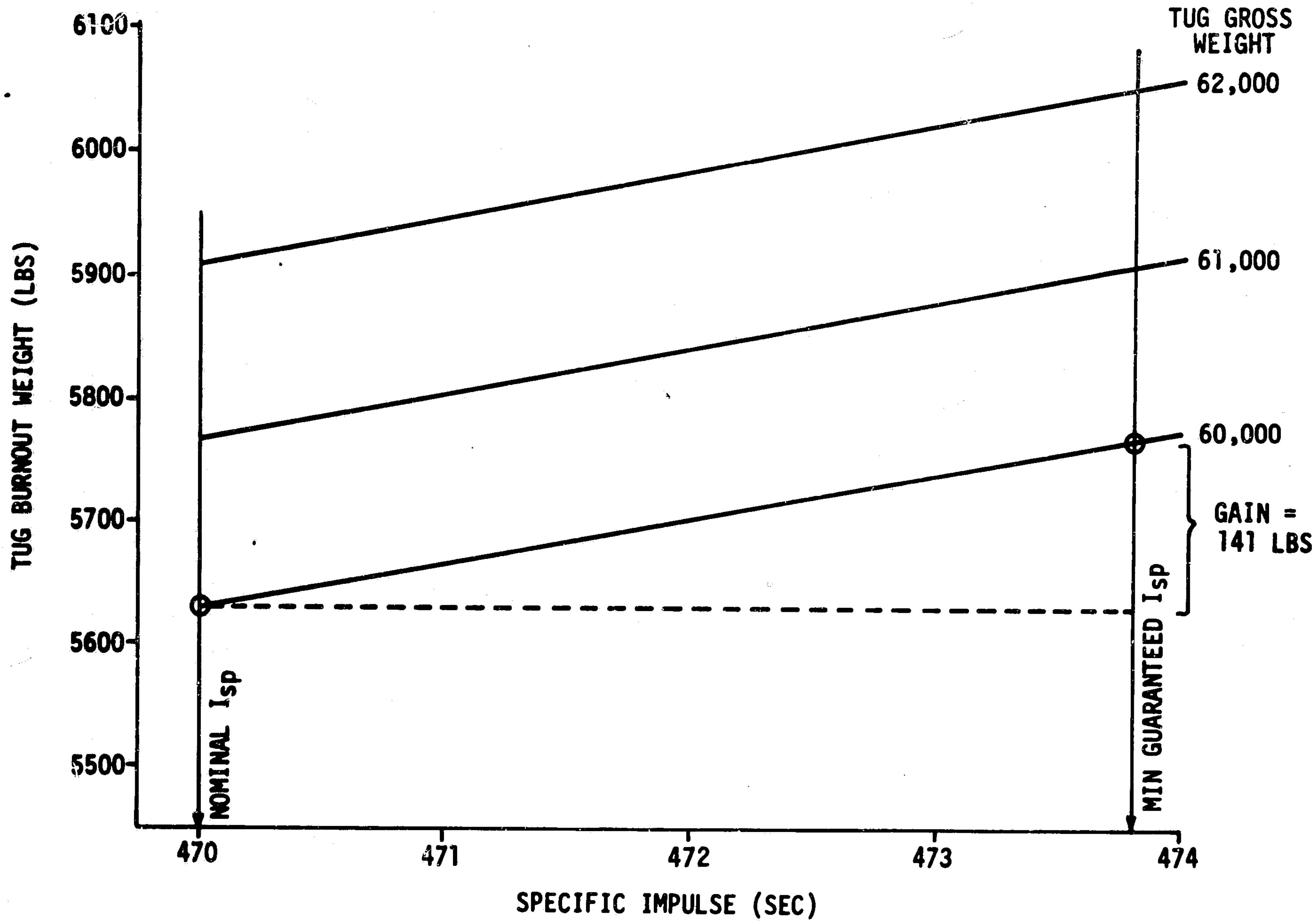


Figure 3.1-7 Effect of Specific Impulse on the Tug Performance

Table 3.1-2 Performance Sensitivity Factors

VARIABLE	FACTOR	VALUE	REMARKS
MASS FRACTION	$\frac{\partial W_{BO}}{\partial \lambda}$	-187.8 LB/.001λ	
SPECIFIC IMPULSE	$\frac{\partial W_{BO}}{\partial I_{SP}}$	37.0 LB/SEC	
PROPELLANT WEIGHT	$\frac{\partial W_{BO}}{\partial W_{PROP}}$	0.16 LB/LB	
NON-PROPULSIVE CONSUMABLES	$\frac{\partial W_{BO}}{\partial W_{CONS}}$	-0.48 LB/LB	THIS IS APPROXIMATE. EXACT VALUE DEPENDS ON WHEN CONSUMABLES ARE DISPOSED
BOILOFF RATE	$\frac{\partial W_{BO}}{\partial W_{BOILOFF}}$	-63.33 LB/LB/HR	
START/STOP LOSS PER BURN	$\frac{\partial W_{BO}}{W_{LOSS}}$	-5.33 LB/LB/BURN	DERIVATIVE ASSUMES 6 MAIN ENGINE BURNS PER MISSION
OUTBOUND ΔV	$\frac{\partial W_{BO}}{\partial \Delta V_{OUT}}$	-.600 LB/FT/SEC	PARTIALS ARE NEARLY EQUAL BECAUSE INBOUND AND OUTBOUND ΔV'S ARE NEARLY EQUAL
INBOUND ΔV	$\frac{\partial W_{BO}}{\partial \Delta V_{IN}}$	-.555 LB/FT/SEC	
MISSION TIME	$\frac{\partial W_{BO}}{\partial T}$	-13.33 LB/DAY	APPROXIMATE. MISSION TIME DEPENDENT
IGNITION WEIGHT	$\frac{\partial W_{BO}}{\partial W_{IGN}}$	0.14 LB/LB	
INBOUND/OUTBOUND	$\frac{\partial W_{PL\ OUT}}{\partial W_{PL\ IN}}$	-2.5 LB/LB	$W_{PL\ OUT} < 3000$ LB
PAYLOAD	$\frac{\partial W_{PL\ OUT}}{\partial W_{PL\ IN}}$	-1.67 LB/LB	$W_{PL\ OUT} > 3000$ LB

3.1.4 Conclusions

The baseline mission requires that the Tug deliver a 3000 lb payload to geosynchronous orbit and return with a second 3000 payload to the shuttle waiting in a 100 n.mi. orbit. This mission requires that the Tug generate a ΔV of 29,000 ft/sec which appears to be extremely demanding in view of the fact that the Tug itself weighs only 5000 to 6000 pounds at burnout. However, the point design study based on 1976 technology indicates that the Tug can do the mission with performance to spare even without resorting to the performance enhancement techniques investigated.

Based on the results of analyses performed in support of the Tug design, the following specific conclusions can be drawn:

1. The NR ΔV budget agrees with the NASA-furnished ΔV budget to within 10 ft/sec.

2. A Tug that can fly the baseline mission successfully can also fly the two alternate missions successfully, provided the ΔV reduction of 215 ft/sec is taken into account.

3. Optimizing the thickness of the singly aluminized kapton insulation (0.5 in.) results in a performance gain of about 25 pounds of allowable Tug burnout weight.

4. Reducing the engine start/stop losses from 15 lb per burn to 10 lb per burn increases the performance capability by about 25 lb of allowable burnout weight.

5. The current NR Tug design can accomplish the baseline mission with performance to spare. The burnout weight or the round trip payload could increase by as much as 720 lb and the Tug could still fly the baseline mission successfully.

3.2 PERFORMANCE ENHANCEMENT TECHNIQUES

Trimming structure weight from the Tug is the most straight-forward way of increasing its performance capability. However, this approach leads to a point of diminishing returns where further gains can only be achieved by using impractical design techniques. Therefore; a more practical approach might be to employ indirect performance enhancement techniques. Presented in Figure 3.2-1 are 16 proposed methods for enhancing the performance capabilities of the Tug.

The four basic approaches to performance improvement are: 1) reducing gravity losses, 2) lowering the ΔV budget, 3) minimizing the non-propulsive consumables weight, and 4) making basic mission changes. The four basic approaches each have four specific techniques designed to improve vehicle

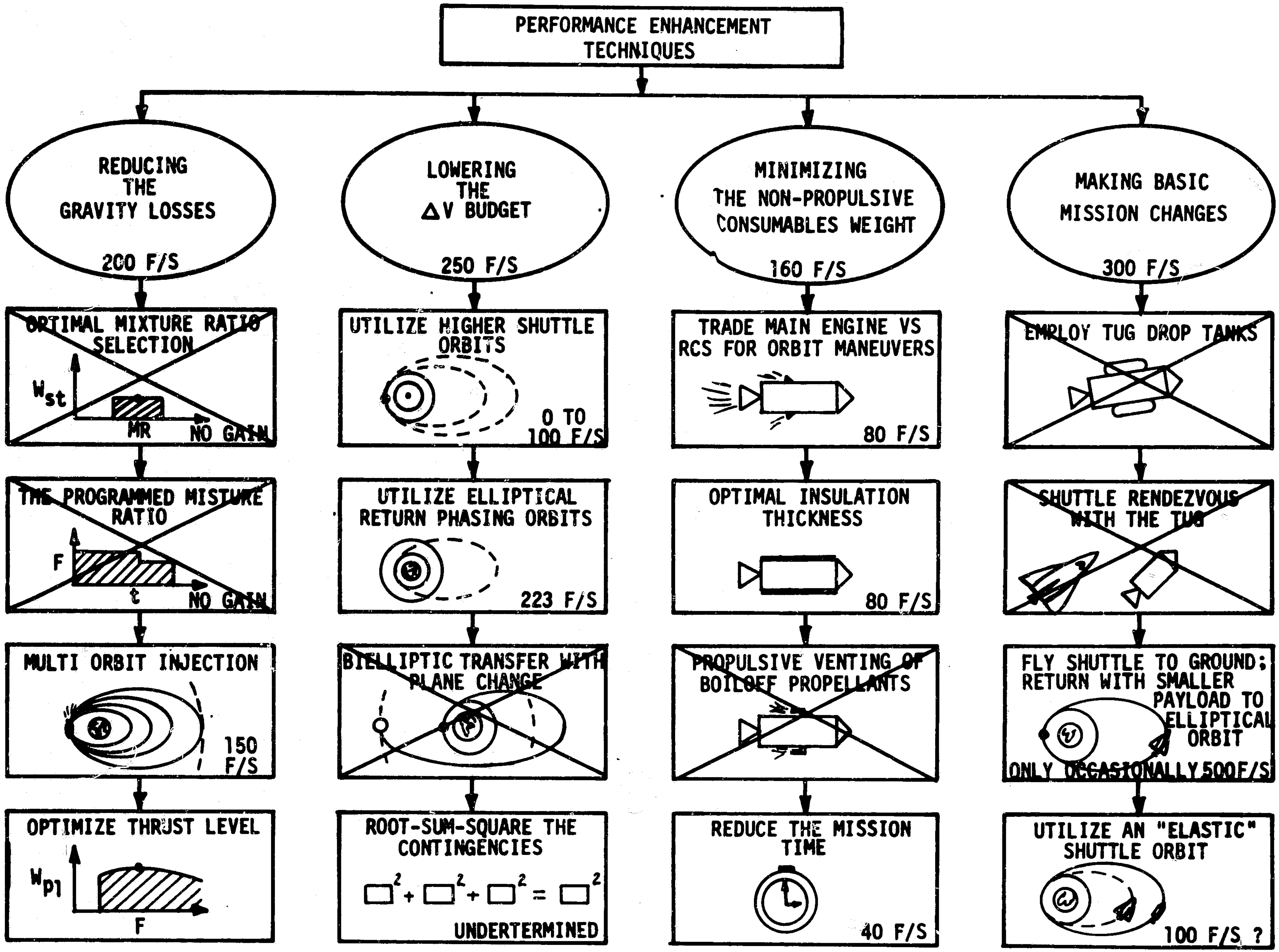


Figure 3.2-1 Performance Enhancement Techniques

performance. A brief analysis indicated that six of these methods were impractical and they were crossed off the list (see Figure 3.2-1). For the ten that remain, estimates of the equivalent expected savings in ΔV are shown in the lower right hand corner of each box. In the following sections, the four most promising performance enhancement techniques are explored in detail.

3.2.1 The Multiorbit Injection Technique

Under normal circumstances the Tug sustains a gravity loss of 310 ft/sec in boosting the payload upward toward the geosynchronous altitude. This gravity loss results from the fact that during its burn the Tug swings outward away from the earth. Consequently, the engines expend added energy in carrying the propellants upward against the pull of gravity. Multiorbit injection is a technique which minimizes these losses.

Multiorbit injection is an alternate method for effectively constraining the altitude of the vehicle during its burn. Essentially, the vehicle's burning program is broken into a series of shorter burns each of which straddles the perigee of a set of elongating transition ellipses. The net result is that all burns are made at the lowest altitude and hence the gravity loss is lower (see Figure 3.2-2). The larger number of burns unfortunately increases the propellant boiloff period and the start/stop losses, thus nulifying part of the potential gain.

Figure 3.2-2 shows what happens to the performance as the number of perigee burns increases. For these simulations all of the burns were of equal duration except for the last one. As seen in the figure, four burns is the optimum and results in a performance gain of 124 pounds in allowable Tug burnout weight. Multiorbit injection thus appears to be a rather attractive performance enhancement scheme which should be given serious consideration for certain suitable Tug applications.

3.2.2 Optional Engine Thrust Level

The mainstage engine of the Tug was groundruled to have a 10,000 lb thrust level. However, increasing the thrust could theoretically yield significant reductions to the 310 ft/sec gravity loss. On the other hand, increasing the thrust would dictate a heavier engine and some of the reduction in gravity losses might not be realizable.

Figure 3.2-3 shows an optimal performance tradeoff which takes these two conflicting factors into account. The optimal thrust level is 12,500 lb and the accompanying gain amounts to about 50 additional pounds of Tug payload. However, any thrust level between 12,000 and 15,000 lb will give approximately the same payload due to the flat maximum section of the curve. Even if the 50 lb payload gain is not required to achieve the performance goals, the switch to a higher thrust level engine is highly recommended. The higher thrust level would make the Tug less sensitive to variations in the initial

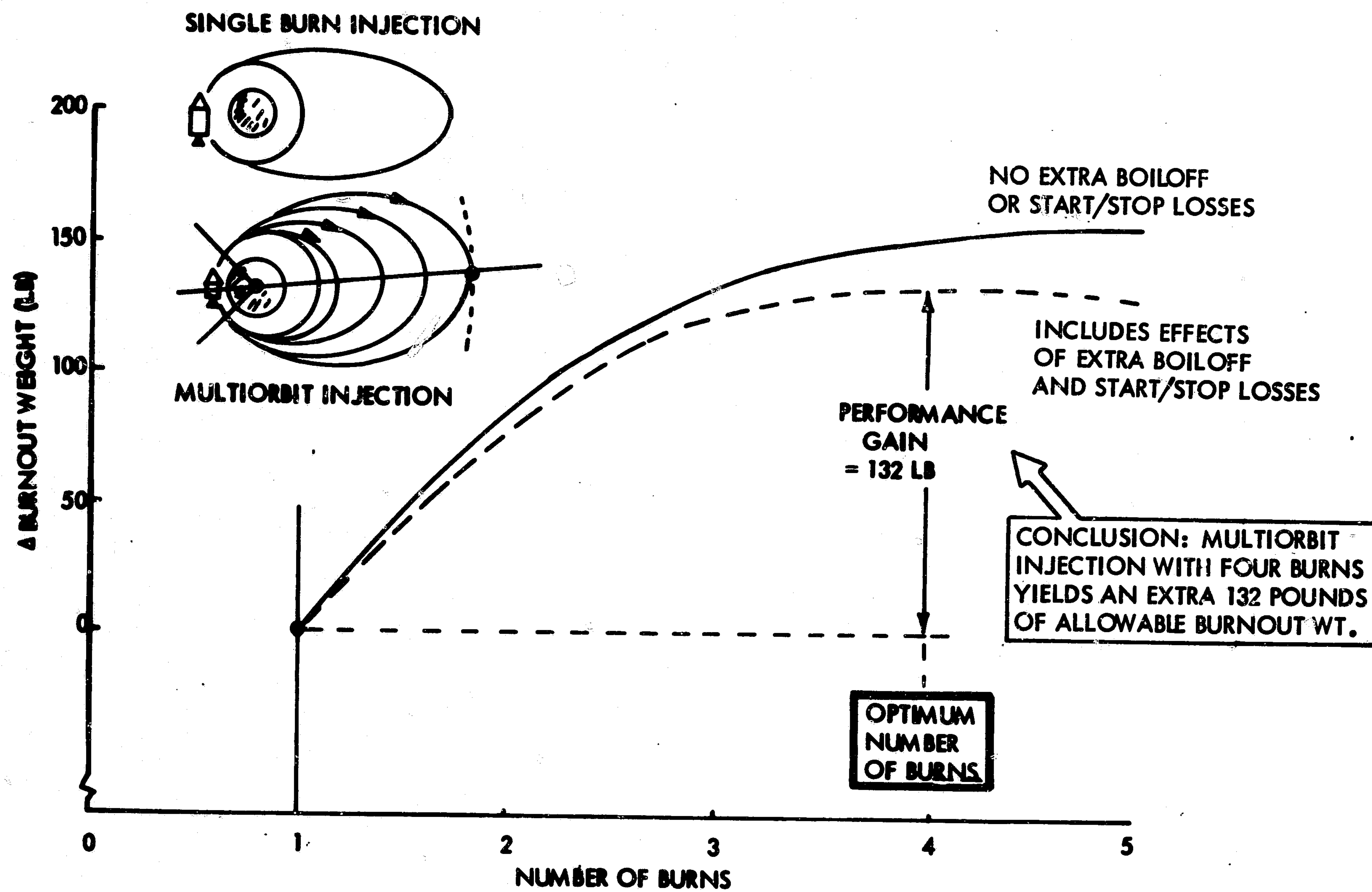


Figure 3.2-2 Multiorbit Injection

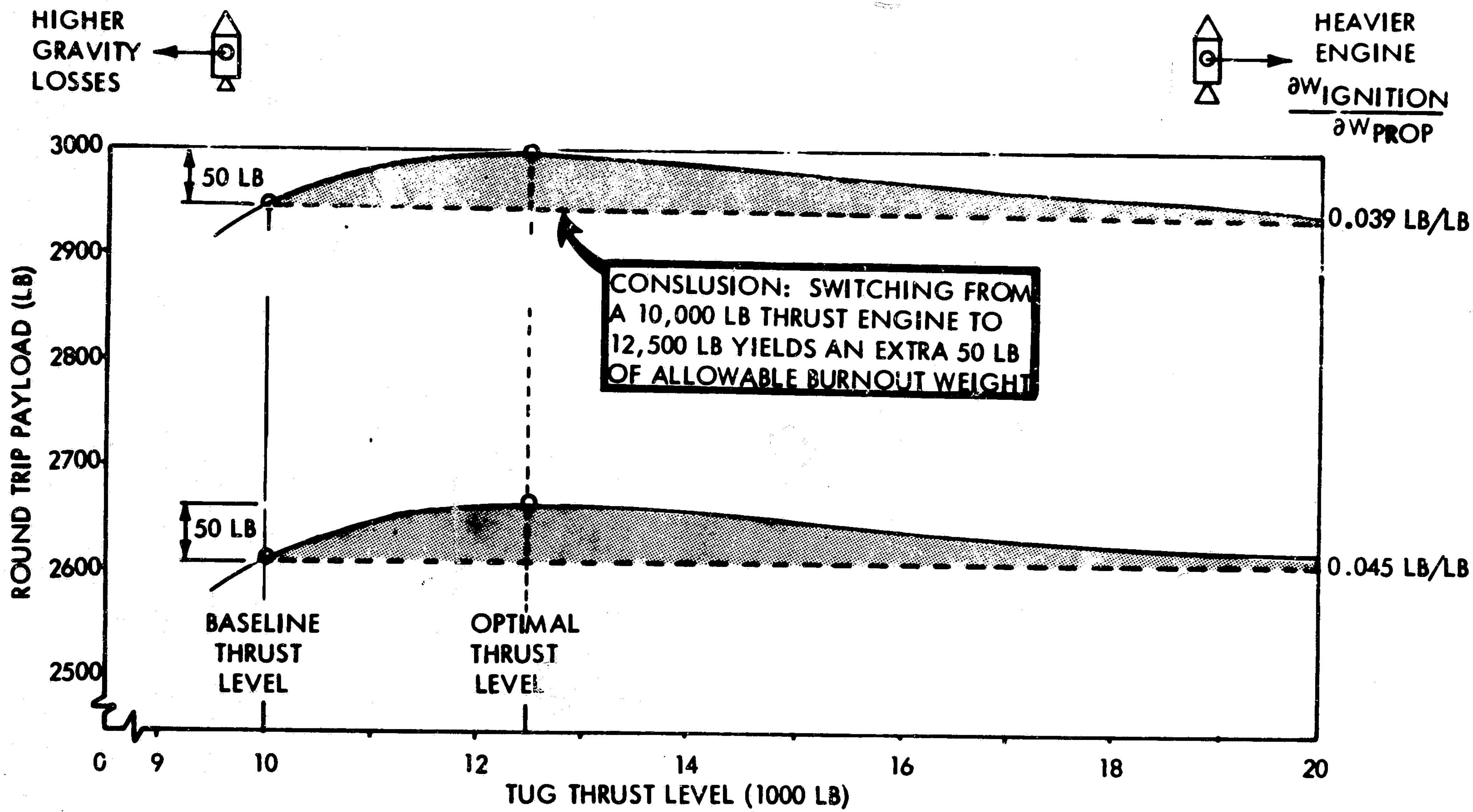


Figure 3.2-3 Thrust Level Optimization

thrust-to-weight ratio. Thus, greater variations in the delivered thrust could be tolerated and the same vehicle would be able to carry heavier payloads without excessive penalties.

3.2.3 The Use of Elliptical Phasing Orbits

According to the mission groundrules, the Tug returns from its orbit at geosynchronous altitude via a 720 n.mi. circular phasing orbit as shown in Figure 3.2-4. The phasing orbit allows the Tug to change its position with respect to the Shuttle (which is in a 100 n.mi. waiting orbit) until it is at the proper point. Once this desired point in space is reached, the Tug executes a Hohmann transfer maneuver and flys to the vicinity of the Shuttle.

The use of circular phasing orbits constitutes a somewhat wasteful approach to phasing. This is true because the first circularization burn is executed at an altitude of 270 n mi rather than at the 100 n mi waiting orbit. Because of the extra altitude some of the intrinsic potential energy in the Tug propellant is wasted and the payload capability of the vehicle is degraded.

This problem can be solved by flying the Tug into an elliptical phasing orbit with a 100 n mi perigee altitude as shown in Figure 3.2-4. Such an elliptical phasing orbit with an apogee of 440 n mi provides the same average phasing rate as the 270 n mi circular phasing orbit yet it reduces the total ΔV budget by approximately 220 ft/sec.

It should be noted, however, that the use of elliptical phasing orbits provides a somewhat lesser degree of mission flexibility. In the case of circular phasing orbits, the burn can be executed at any point along the orbit. On the other hand, with elliptical phasing orbits, the circularization burn can be made only at the 100 n mi level, i.e., at perigee. However, flexibility can be introduced into the elliptical phasing orbit approach by allowing the apogee altitude of the phasing orbit to vary depending upon where the Shuttle is located at the time the Tug performs its first braking maneuver. As is shown in Figure 3.2-4, a slight penalty can result from varying the apogee altitude but it amounts to only 2 ft/sec for every 100 n mi the apogee is increased. In view of the overall gain resulting from use of elliptical phasing orbits, this small penalty is certainly tolerable. The penalty stems from the fact that the plane change accompanying the first two burns is accomplished in a less efficient manner when the magnitude of the burn is decreased.

Elliptical phasing orbits yield large performance gains with no significant disadvantages. Therefore, it is recommended that elliptical phasing orbits be considered for future Tug performance studies.

3.2.4 Reducing the Mission Duration

The six day duration of the baseline mission appears to be conservative in various respects. For example, a 3 day interval was allotted for the first phasing/rendezvous sequence when 2 days would apparently suffice. If the

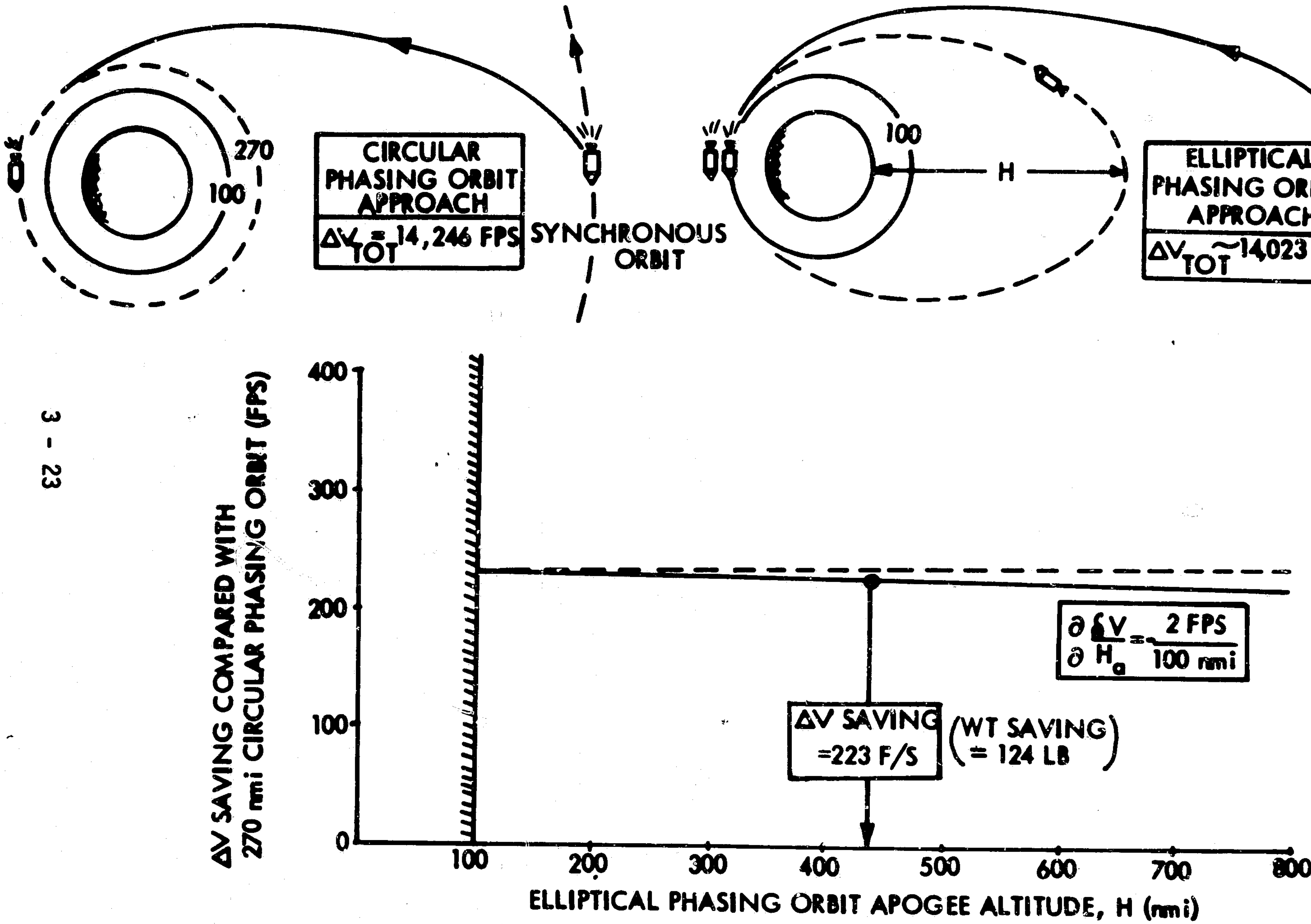


Figure 3.2-4 Elliptical Phasing Orbits

mission duration could be decreased to reduce the propellant boiloff interval, the performance capability of the Tug could be increased. As indicated in Figure 3.2-5, small performance gains can be obtained in this manner. For example, if a one pound per hour boiloff rate is assumed, it turns out that shortening the mission duration from 6 to 4 days would increase the allowable Tug burnout weight by about 30 pounds.

3.2.5 Conclusions

Based on the results of the performance enhancement analysis the following conclusions can be drawn:

1. The baseline thrust level of 10,000 lb should be increased to at least 12,500 lb. Raising the thrust to this level increases the payload capability by 50 lb and would make the Tug less sensitive to thrust level excursions and payload weight variations.

2. The multiorbit injection technique with 4 perigee burns increases the allowable Tug burnout weight (or roundtrip payload) by 132 lb over the single burn case.

3. Switching from a 270 n mi circular phasing orbit to an elliptical phasing orbit with an apogee altitude of up to 100 n mi would decrease the required ΔV budget by about 220 ft/sec (124 lb).

3.3 PERFORMANCE STATUS OF THE TUG

The current performance status of the Tug is summarized in Figure 3.3-1. Note that the baseline vehicle can grow by 720 lb and still fly the baseline mission successfully. This performance pad can be further increased by approximately 450 pounds provided the following steps are taken:

1. Increase the main engine thrust level to 12,500 lb.

2. Use baseline main engine nominal specific impulse instead of minimum guaranteed.

3. Utilize the multiorbit injection technique.

4. Employ elliptical phasing orbits.

In addition, a further gain could be made as follows:

1. Reduce the mission duration from 6 to 4 days.

2. Employ optimal assembly orbits.

However, these two possibilities have not at this time been studied in sufficient detail to warrant their recommendation.

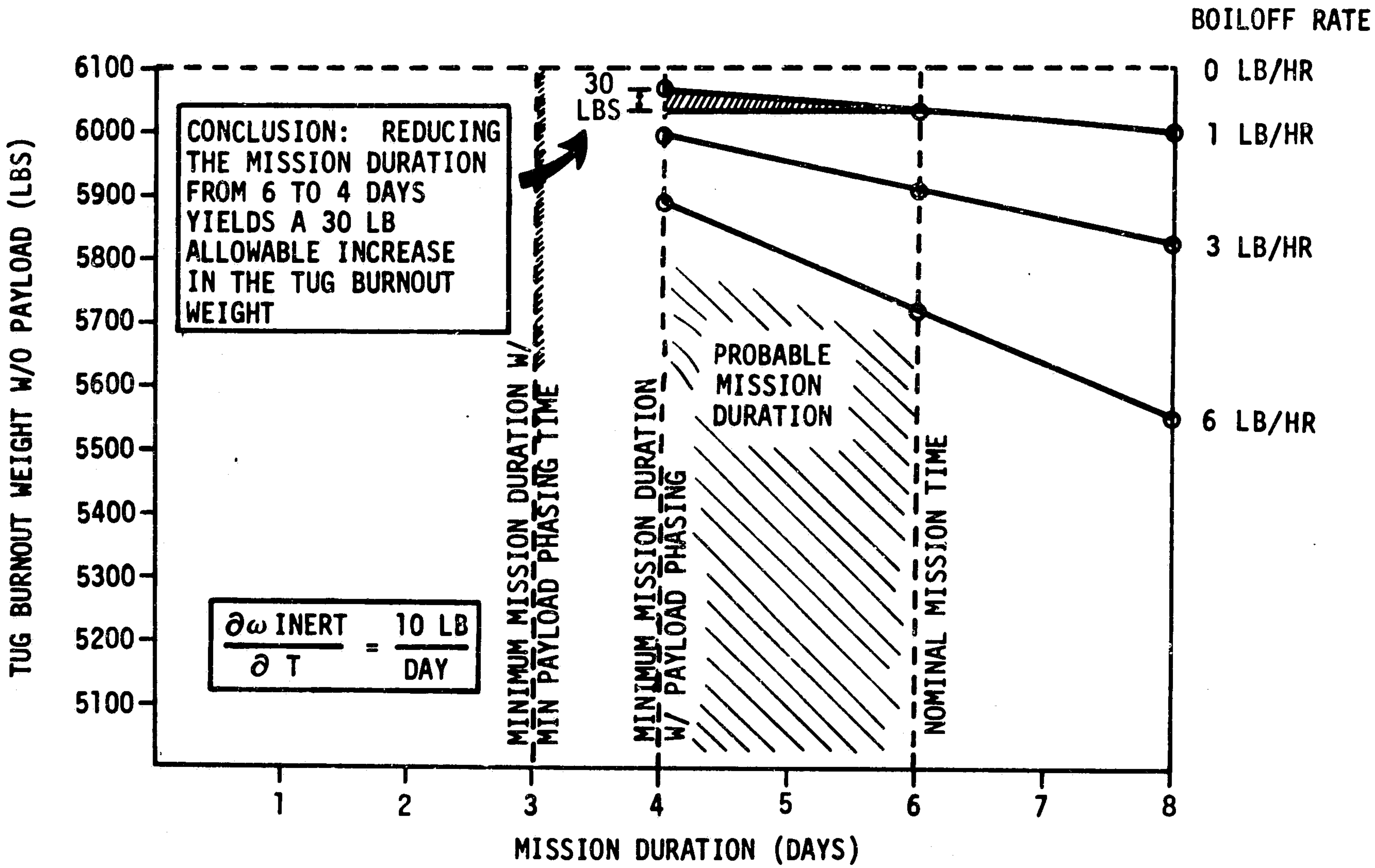
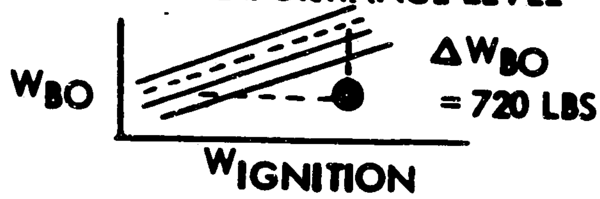


Figure 3.2-5 Effect of Mission Duration on the Tug Performance

BASELINE PERFORMANCE LEVEL



PERFORMANCE ENHANCEMENT TECHNIQUES

TECHNIQUE	POTENTIAL GAIN ΔW_{BO} (LB)
OPTIMUM THRUST	50
MULTI ORBIT INJECTION	132
ELLIPTICAL PHASING ORBITS	124
MISSION DURATION REDUCTION	30
OPTIMAL ASSEMBLY ORBIT ALTITUDE	TBD

POINT DESIGN DECISIONS

TECHNIQUE	POTENTIAL GAIN ΔW_{BO} (LB)
OPTIMAL INSULATION	20
REDUCE START/STOP LOSSES	25
ΔV BUDGET ALLOCATION	VAR

Figure 3.3-1 Baseline Performance Summary

4.0 FLIGHT REQUIREMENTS

Design of the Tug structure and the various subsystems is dependent upon many requirements established in the flight technology area. The three general areas in which flight requirements are generated are; 1) aerothermodynamics, 2) thermal control and protection systems and 3) guidance, navigation and control. Flight requirements established in these areas are presented in the following sections. Tug compartment and insulation venting requirements resulting from parametric studies are discussed in Section 4.1. Thermal protection requirements for the propellant tanks, the avionics system, the auxiliary propulsion system and the Tug base region are presented in Section 4.2. Requirements for guidance and navigation, the thrust vector control (TVC) system and the auxiliary propulsion system (APS) are provided in Section 4.3.

4.1 AEROTHERMODYNAMICS

Both venting and plume impingement were investigated in Aerothermodynamics. A parametric study was made to obtain pressure time histories for the Tug compartment venting and the aluminized mylar insulation venting. Analyses were also performed to determine radiation for the main engine plume and the effect of RCS engine plume impingement on the main engine.

4.1.1 Tug Compartment and Insulation Venting

Just prior to launch, the non-pressurized compartments and cavities of the vehicle are at atmospheric pressure (or greater due to purge gas pressure distribution). During boost, the local external pressure drops rapidly and the internal pressure in the compartment lags. The resulting differential pressure causes a bursting or crushing load on the compartment wall depending on the location of the compartment and the shape of the vehicle. The size and location of external vent orifices and inter-compartment vents must be chosen to limit the compartment wall differential pressures to acceptable values, depending on the structural design criteria.

A parametric study was conducted to select the vent areas for the Tug compartments and determine the differential pressure across the external skin and the pressure inside the multi-layer insulation (MLI) of the Tug during venting. The study was made using the boost and entry trajectories obtained from the NR Shuttle project. The NR multiple chamber venting program was utilized for the analysis.

The venting program uses the internal gas conditions within the chamber and the external local properties (i.e., velocity, temperature and pressure) to compute the mass flow through the vent. Starting with the initial conditions, the time history of the compartment pressure is obtained. The difference between the compartment internal pressure and the local pressure just outside the vent was obtained for a range of ratios of vent area to compartment volume.

The results of the venting study are presented in three parts: 1) boost phase, 2) entry phase, and 3) insulation venting. Throughout the study an insulated system is assumed with no heat transfer between the compartment wall and the gases or between the Shuttle Orbiter compartments and the Tug compartment. This assumption may affect the differential pressure across the compartment walls.

Boost Phase Compartment Venting

During the boost phase of the mission, the Tug is carried inside the cargo bay of the Shuttle Orbiter. Since the Tug remains inside the cargo bay during ascent, the Shuttle internal pressure is the controlling factor in the Tug compartment venting. The study was concerned with the selection of the vent areas for the Tug compartment which would minimize the differential pressures across the Tug external skin. The external local properties for the cargo bay compartment were obtained from the Shuttle/Tug boost trajectory data shown in Figure 4.1-1, where the ambient pressure was used for the local pressure outside the vent opening. In order to help select the vent area, various area-to-volume ratios were used (0.001 to 0.25 square inches per cubic foot of free volume.) Figures 4.1-2 and 4.1-3 show the maximum differential pressure obtained for various area-volume ratios.

In order that the Tug internal pressure would follow cargo bay pressure very closely, an area-volume ratio of 0.06 in^2/ft^3 was selected. The internal pressure history of the Tug free volume is shown in Figure 4.1-4 using the area-volume ratio of 0.06 in^2/ft^3 (which is the same area volume ratio as for the cargo bay compartment). For the selected area-volume ratio, the pressure histories and the maximum differential pressure (burst loading) for the Tug were obtained. Figure 4.1-5 presents the internal pressure for the Tug and the cargo bay compartment internal and external pressures, while Figure 4.1-6 shows the maximum differential pressures on both the Tug compartment and the cargo bay compartment. Figure 4.1-7 presents the last portion of the ascent phase. The data are presented in Figure 4.1-7 in terms of PSF and Torr for the purpose of using this as initial pressure in the mylar insulation venting. For the boost phase, the results show that the Tug outer skin is subjected to a bursting load of 0.071 PSID.

Entry Phase Compartment Venting

During the entry phase, the Tug enters the atmosphere while inside the Shuttle cargo bay. The entry venting analysis was made using the Shuttle/Tug entry trajectory data shown in Figures 4.1-8 and 4.1-9. Since the Tug enters the atmosphere inside the Shuttle cargo bay, the inward venting is controlled by the cargo bay compartment internal pressure. The cargo bay as well as the Tug compartment is subjected to a constant crushing load throughout the entry phase. Since the area-volume ratio chosen for the Tug will allow the internal pressure to follow closely the cargo bay compartment internal pressure, the crushing load on the Tug external skin is very small. Figure 4.1-10 shows the maximum differential pressure experienced by both the Tug and the cargo bay compartment. The results show that the maximum crushing load on the Tug outer skin is 0.0008 PSID.

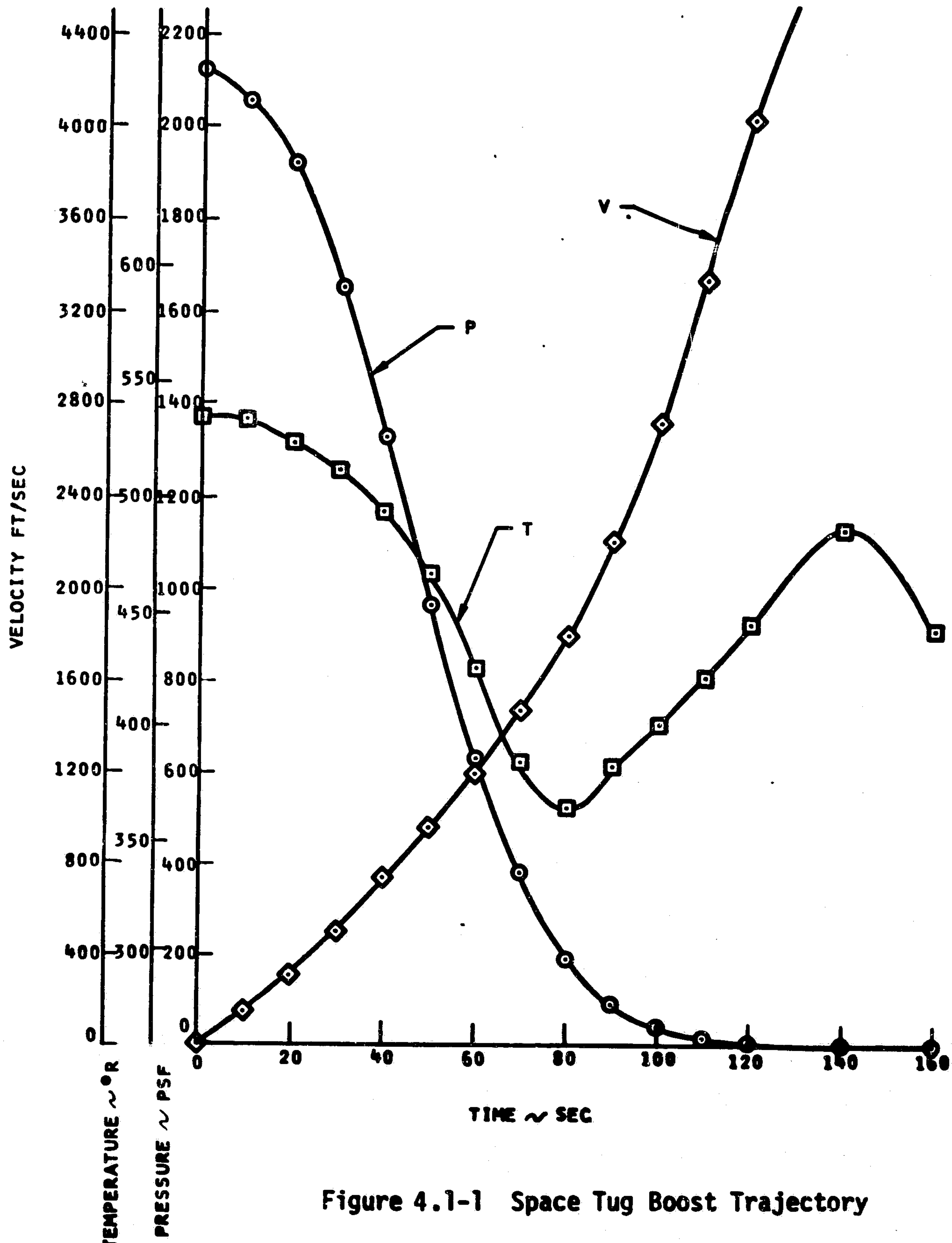


Figure 4.1-1 Space Tug Boost Trajectory

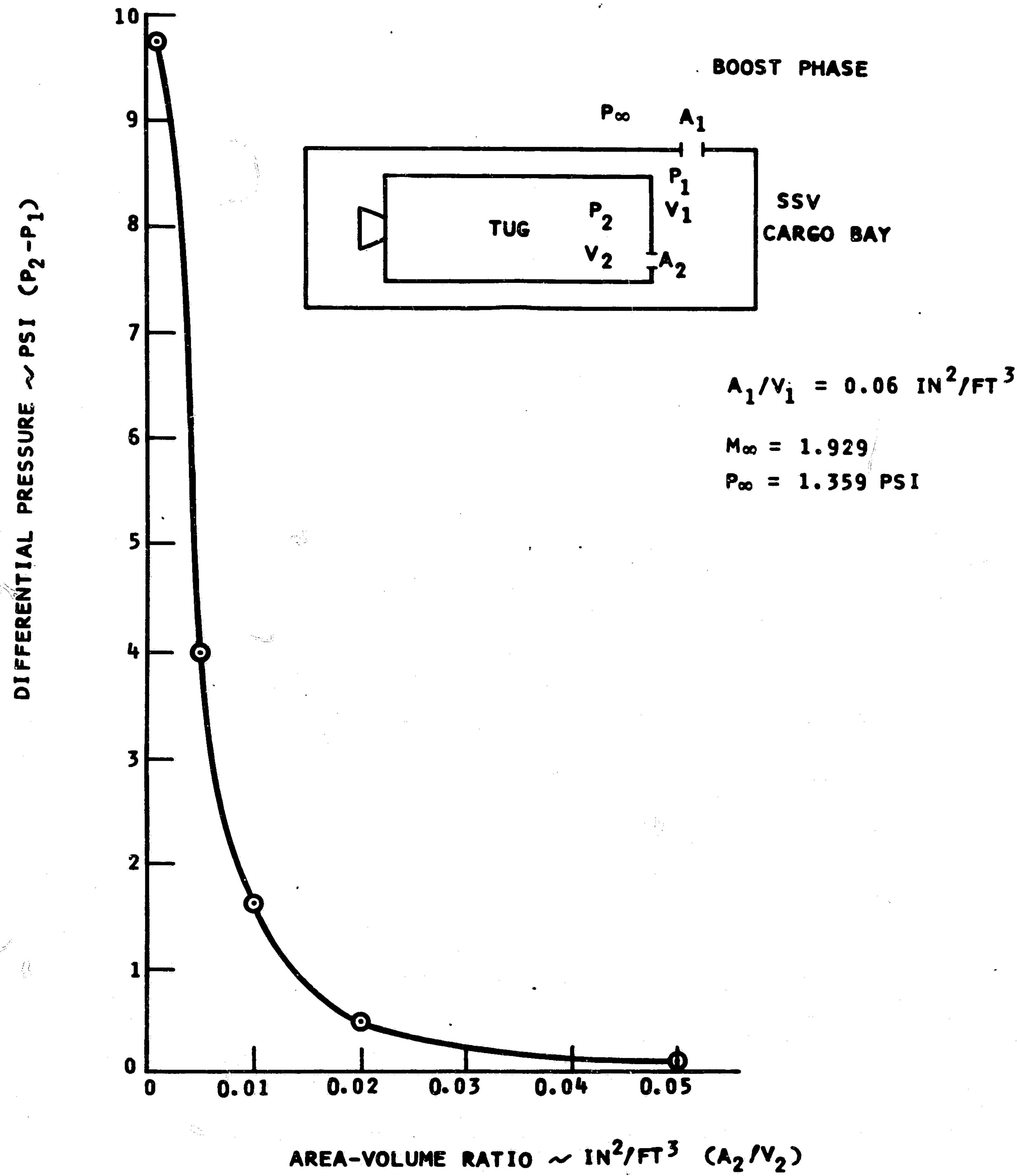


Figure 4.1-2 Differential Pressure Across Tug Outer Skin Boost Phase

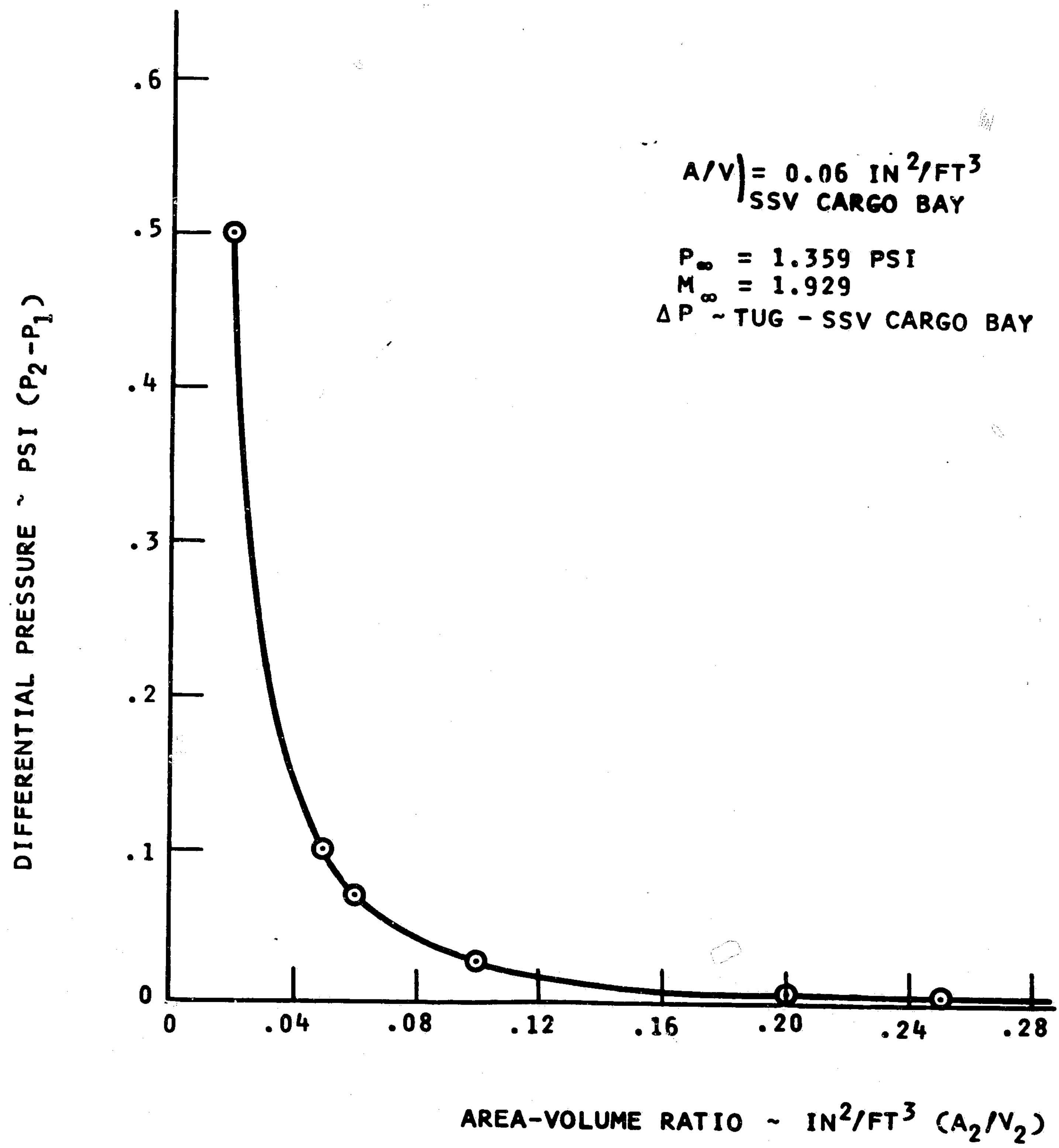


Figure 4.1-3 Differential Pressure Across Space Tug Outer Skin

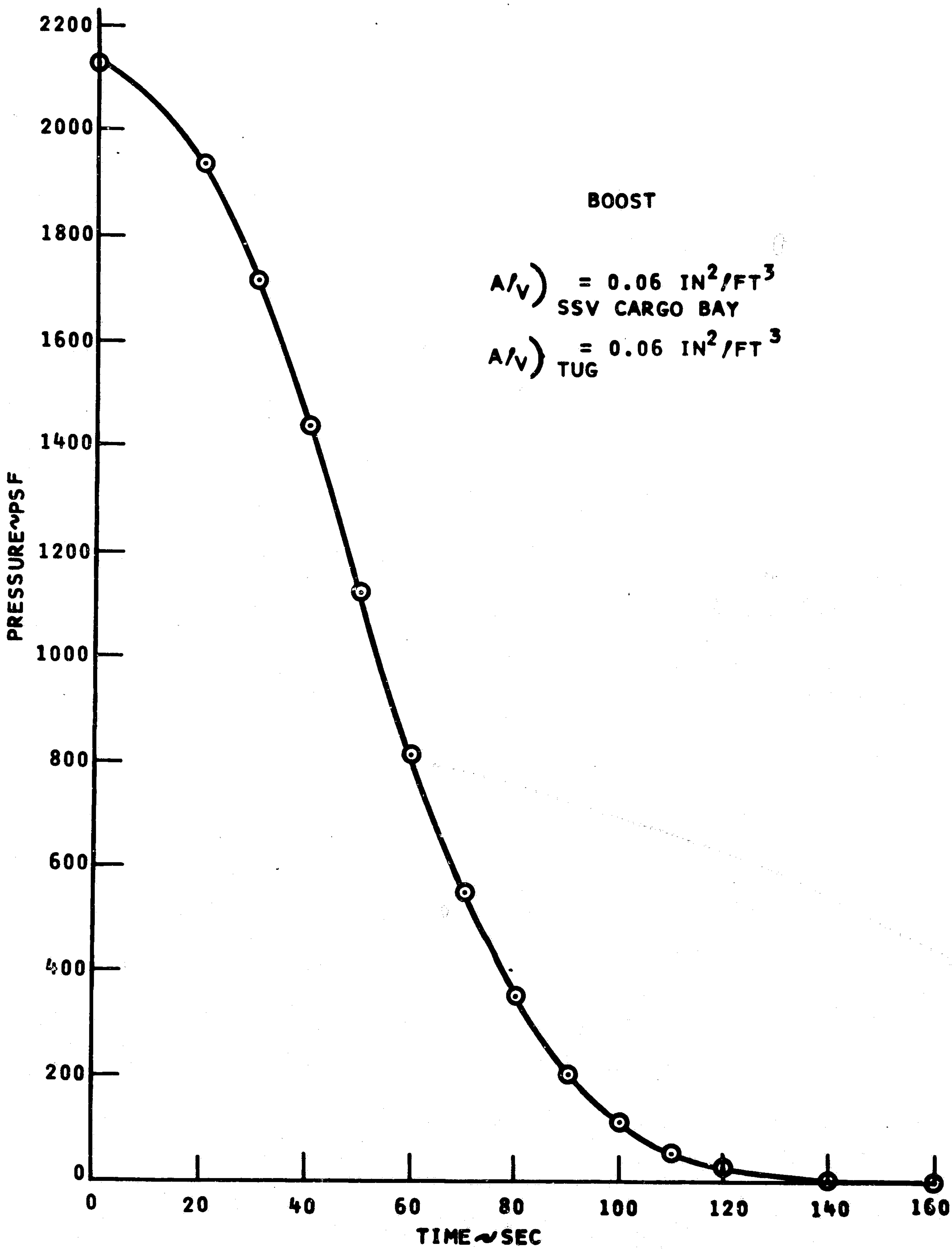


Figure 4.1-4 Space Tug Free Volume Internal Pressure

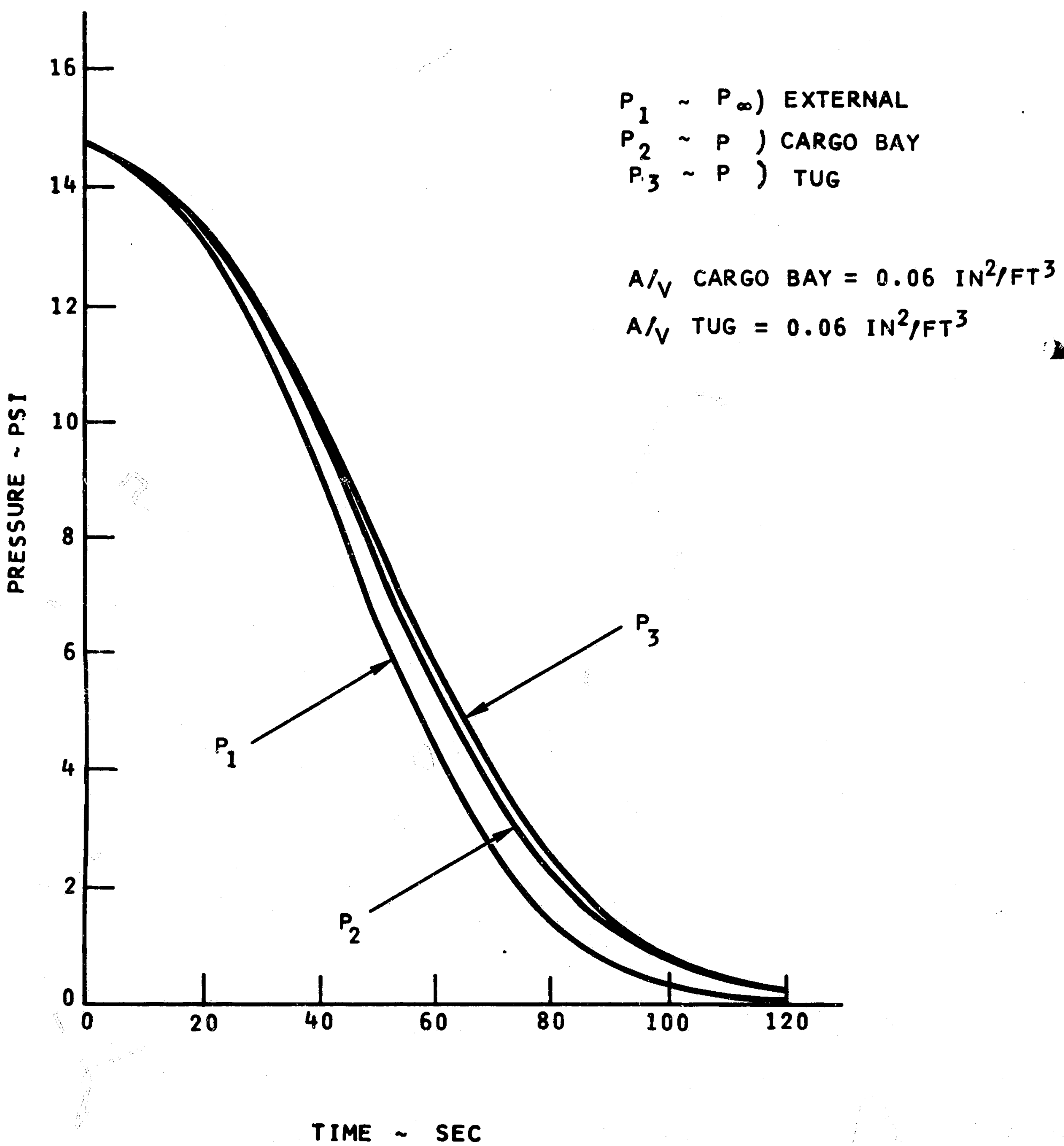


Figure 4.1-5 Space Tug Pressures History

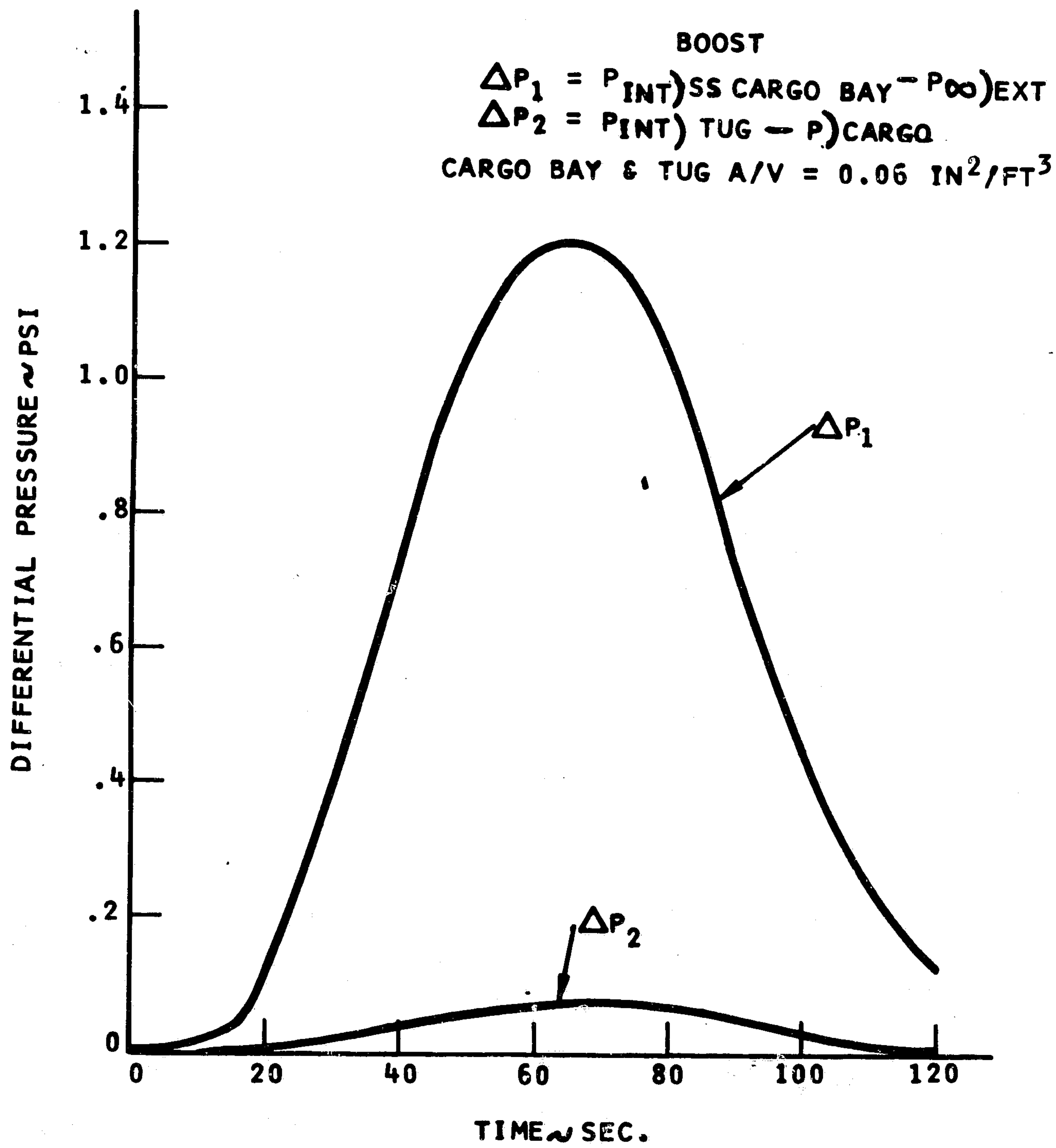


Figure 4.1-6 Space Tug Differential Pressure

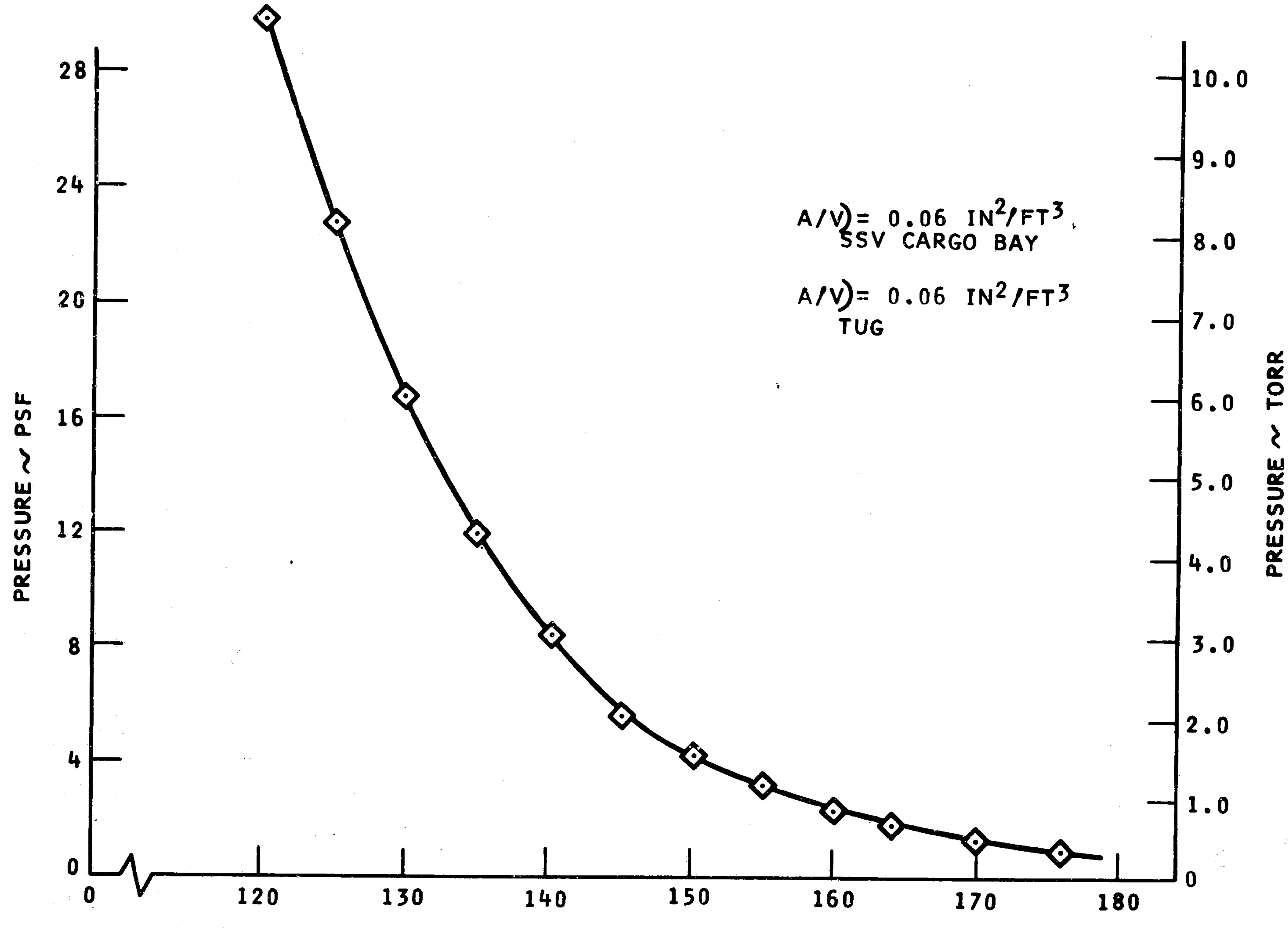


Figure 4.1-7 Space Tug Internal Pressure ~ Free Volume Boost

Insulation Venting

The time required to boost the Tug (inside the Shuttle cargo bay) into a 100 n.mi. orbit is approximately 51 minutes. One orbit later (120 minutes) the Tug is separated from the Shuttle and boosts to a geosynchronous orbit (19,300 n.mi.). The LH_2 and LOX tanks inside the Tug structural frame are insulated with a crinkled aluminized multi-layer mylar blanket about 0.5 inches thick. The mylar layers are stacked at 60 layers per inch. The trapped helium between the mylar layers requires time to vent to the atmosphere (by way of the Tug compartment). After the first orbit, it is assumed that the pressure inside the insulation has reached 1 Torr. Since it is desirable to have the pressure between the layers drop rapidly to 10^{-4} Torr or less for thermal purposes, a calculation has been made to determine the amount of time required for the pressure to drop to the desired level. In calculating the time history for the pressure drop within the insulation, it was assumed that no pressure gradient exists between two adjacent layers of mylar and that there is no pressure lag from the butt end of the blanket to the pressure in the Tug compartment. From wind tunnel data, Figure 4.1-11 presents the outgassing rate for an individual insulation composite of crinkled aluminized mylar versus pumping time. The mylar insulation venting pressure time history was obtained using the outgassing rate and the initial pressure (1 Torr) inside the Tug compartment at 100 n.mi. orbit. The results are presented in Figures 4.1-12 and 4.1-13. It can be seen that approximately 345 minutes are required for the pressure within the insulation layers to drop to 10^{-4} Torr. The above result was obtained using outgassing rates taken at 540°R temperature and 60 layers per inch of mylar. Figure 4.1-14 shows the multilayer spacing effect on the insulation venting from which it can be seen that the lesser the number of layers per inch, the faster the pressure drop. Figure 4.1-15 presents the effect on pressure due to the variation of temperature. For example, using the 60 layers per inch blanket, a drop in temperature of about 70°R results in an order of magnitude drop in pressure.

Conclusion

In conclusion, for the area-volume ratio of 0.06 in^2/ft^3, no problems are anticipated in the Tug venting during boost and entry. However, the mylar blanket insulation venting time required to drop to 10^{-4} Torr was a rough estimation based on a simplified analysis using experimental outgassing data. A better definition of the insulation criteria should be established before any detailed analysis is made on the blanket venting.

4.1.2 Main Engine and RCS Exhaust Plume

The single main engine exhaust plume induces radiative heating to the Tug base region, while the pitch and roll RCS engine plumes impinge on the Tug surface producing appreciable local convective heating rates. The main engine and RCS exhaust plume properties were determined by means of a method of characteristics computer program. The right-running characteristic start line for the plume program was generated by an approximate method which is based on the nozzle lip angle and Mach number. Thermodynamic properties were computed by means of the NASA/Lewis propellant analysis program down to

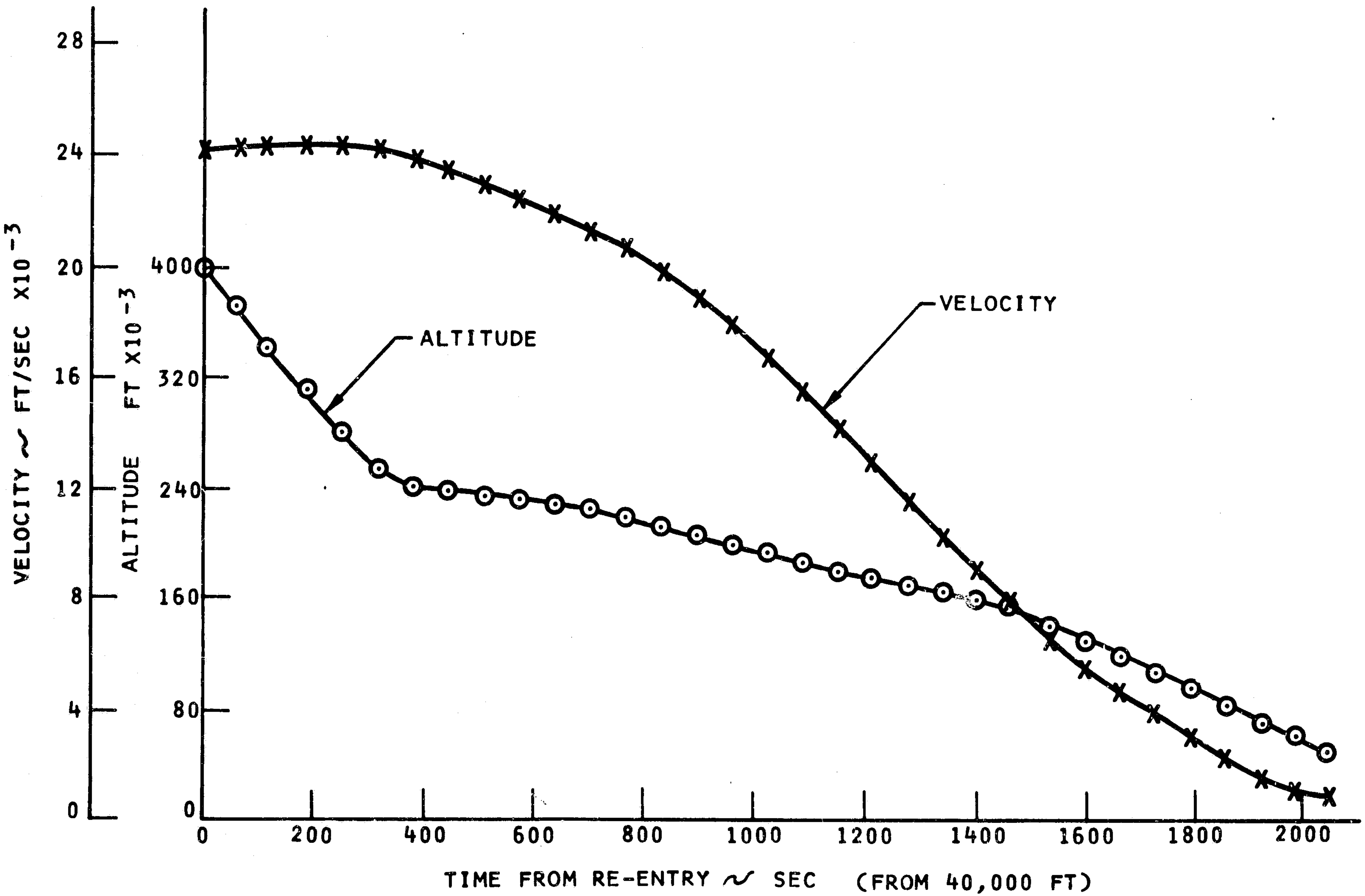


Figure 4.1-8 Space Tug Re-Entry Trajectory

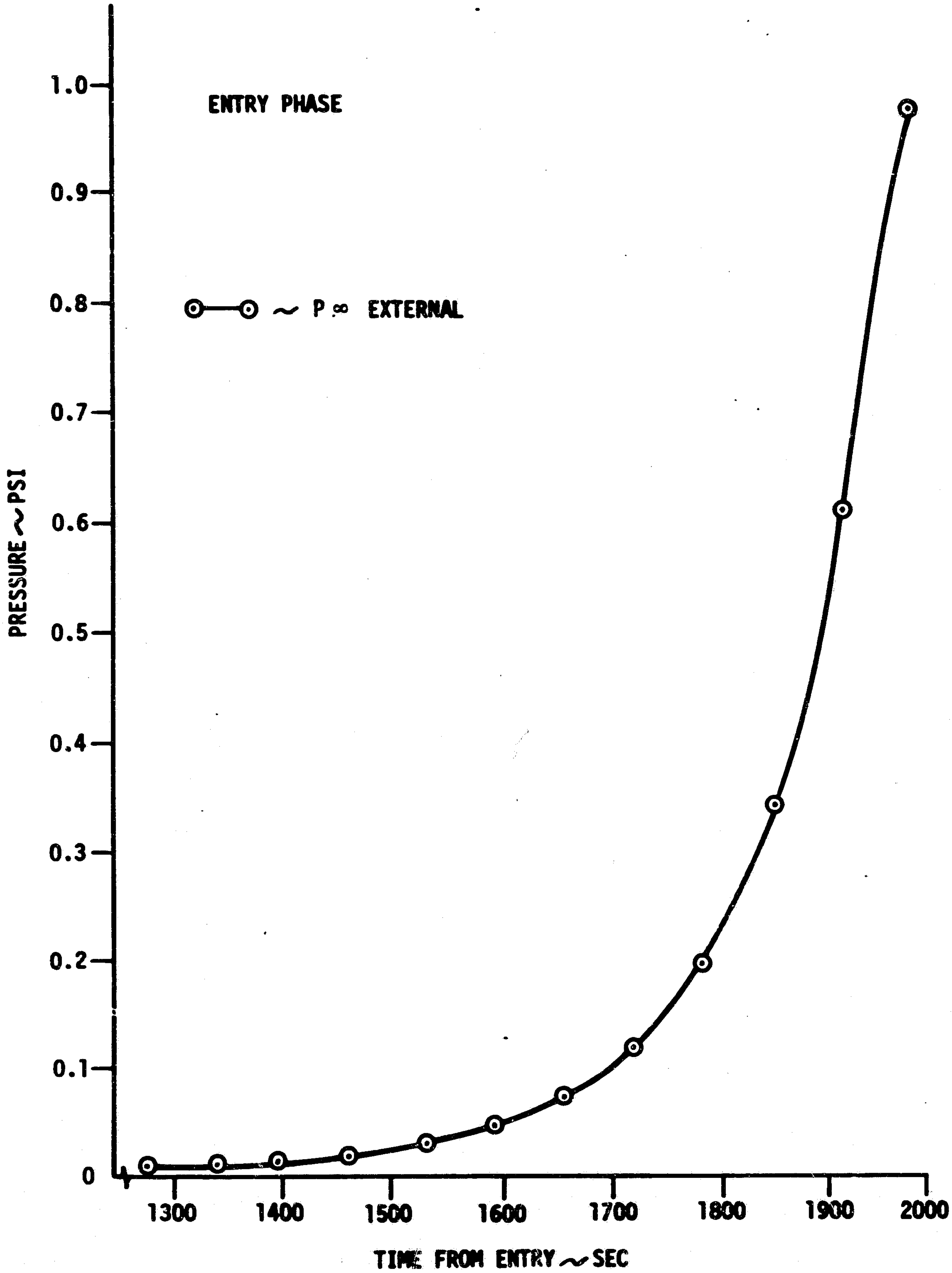


Figure 4.1-9 Space Tug Pressure History

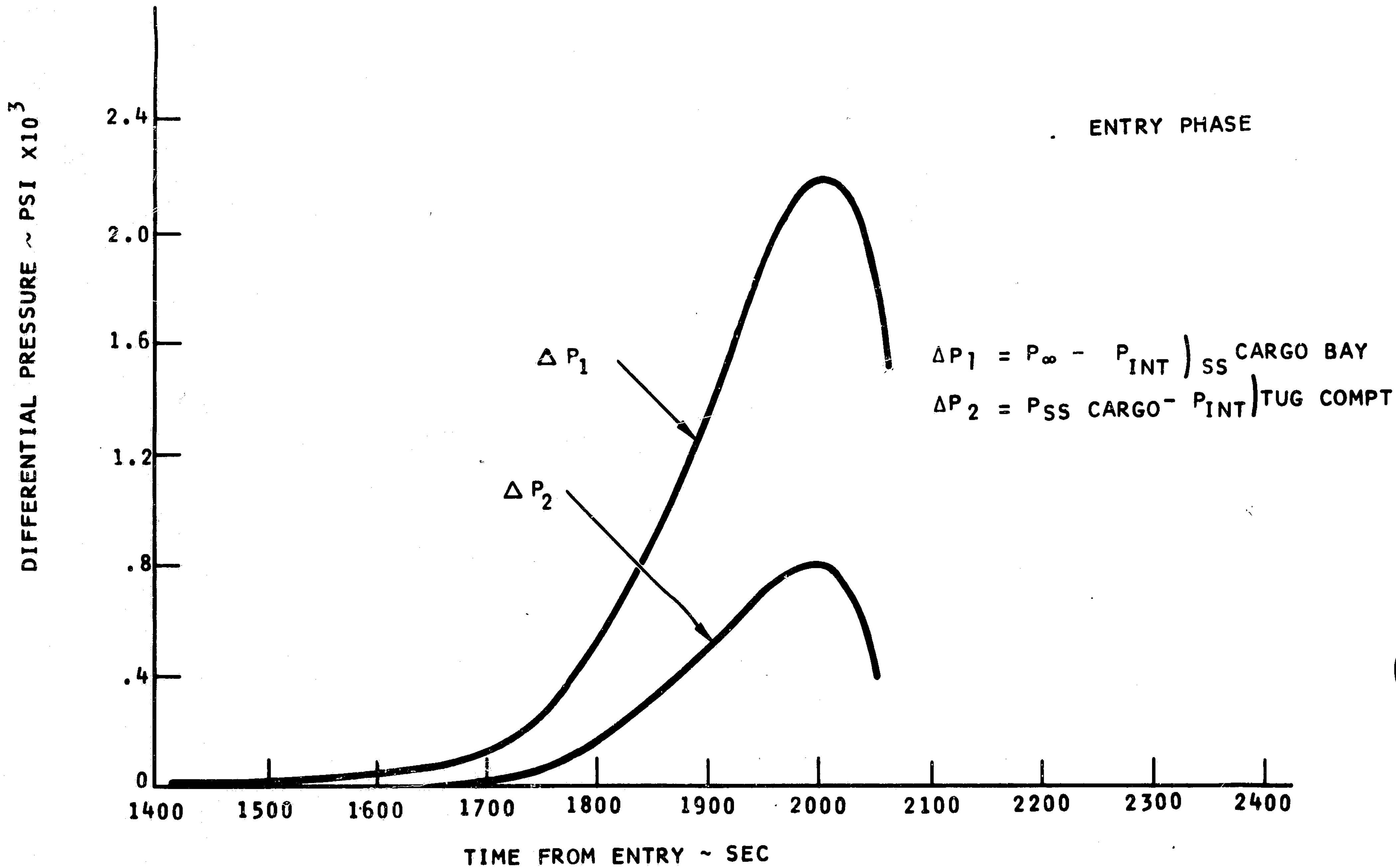


Figure 4.1-10 Space Tug Differential Pressure

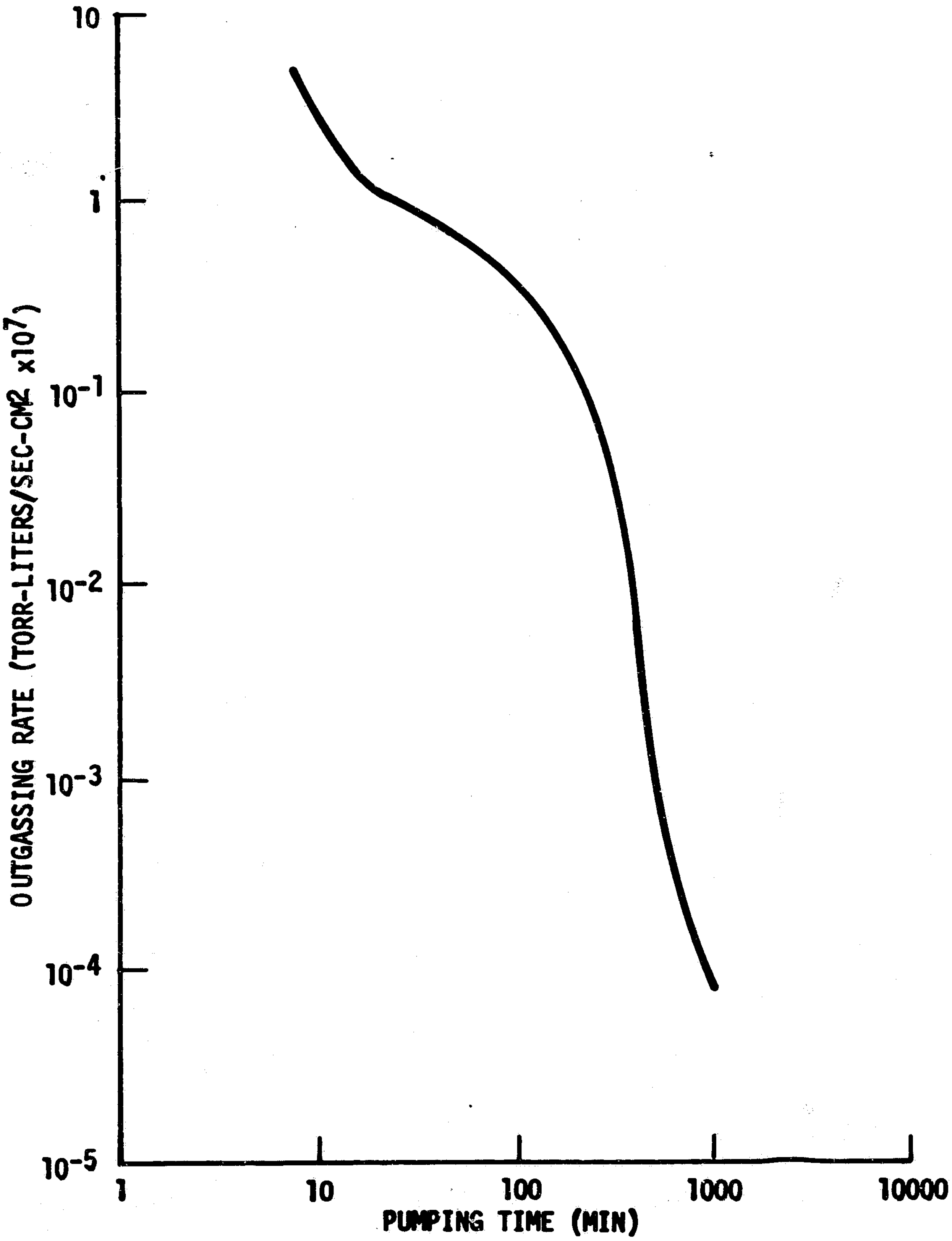


Figure 4.1-11 Outgassing Rate for Individual Insulation Composite Crinkled Aluminized Mylar 0.25-MIL @ 540°R

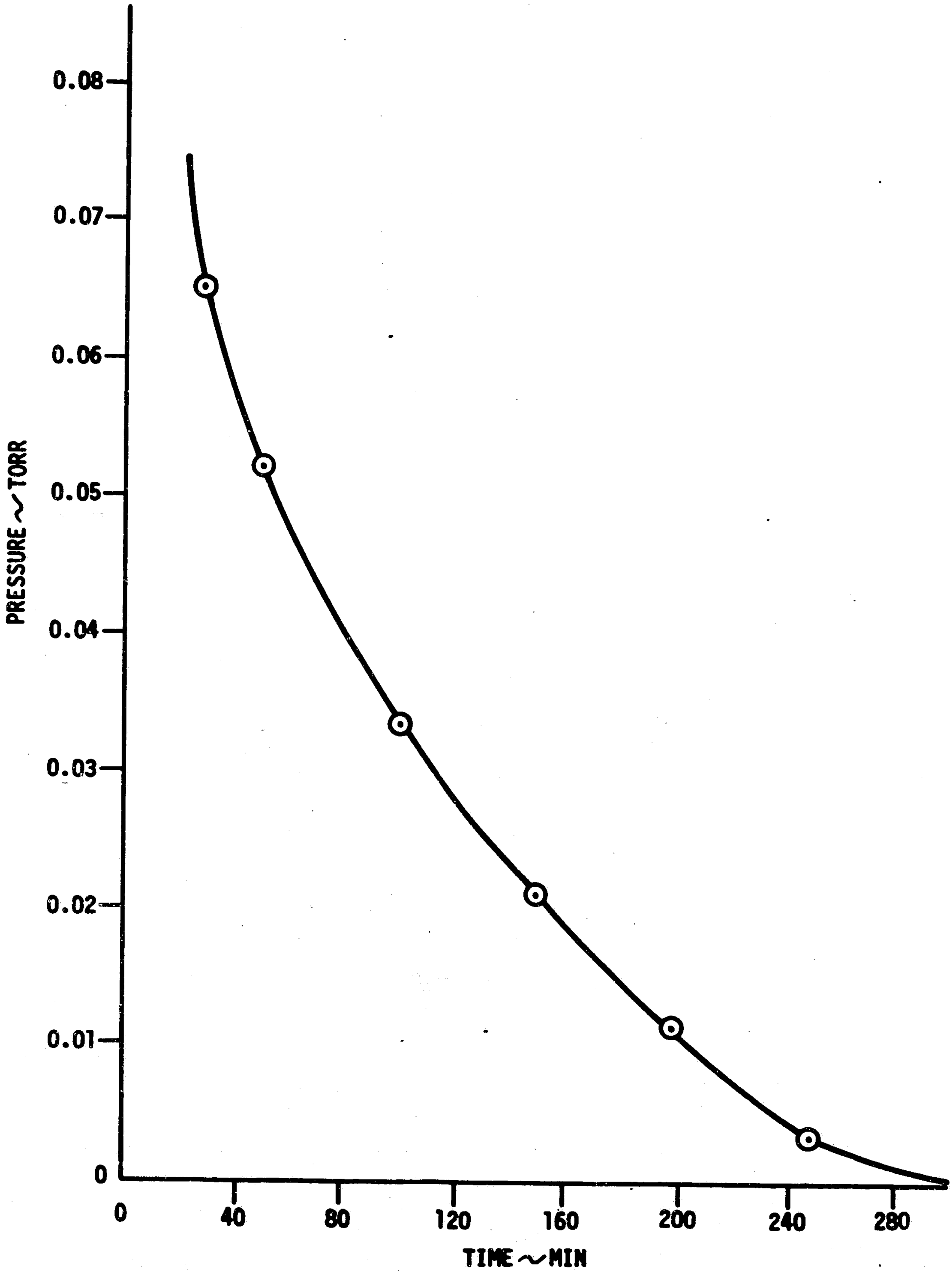


Figure 4.1-12 Mylar Insulation Venting

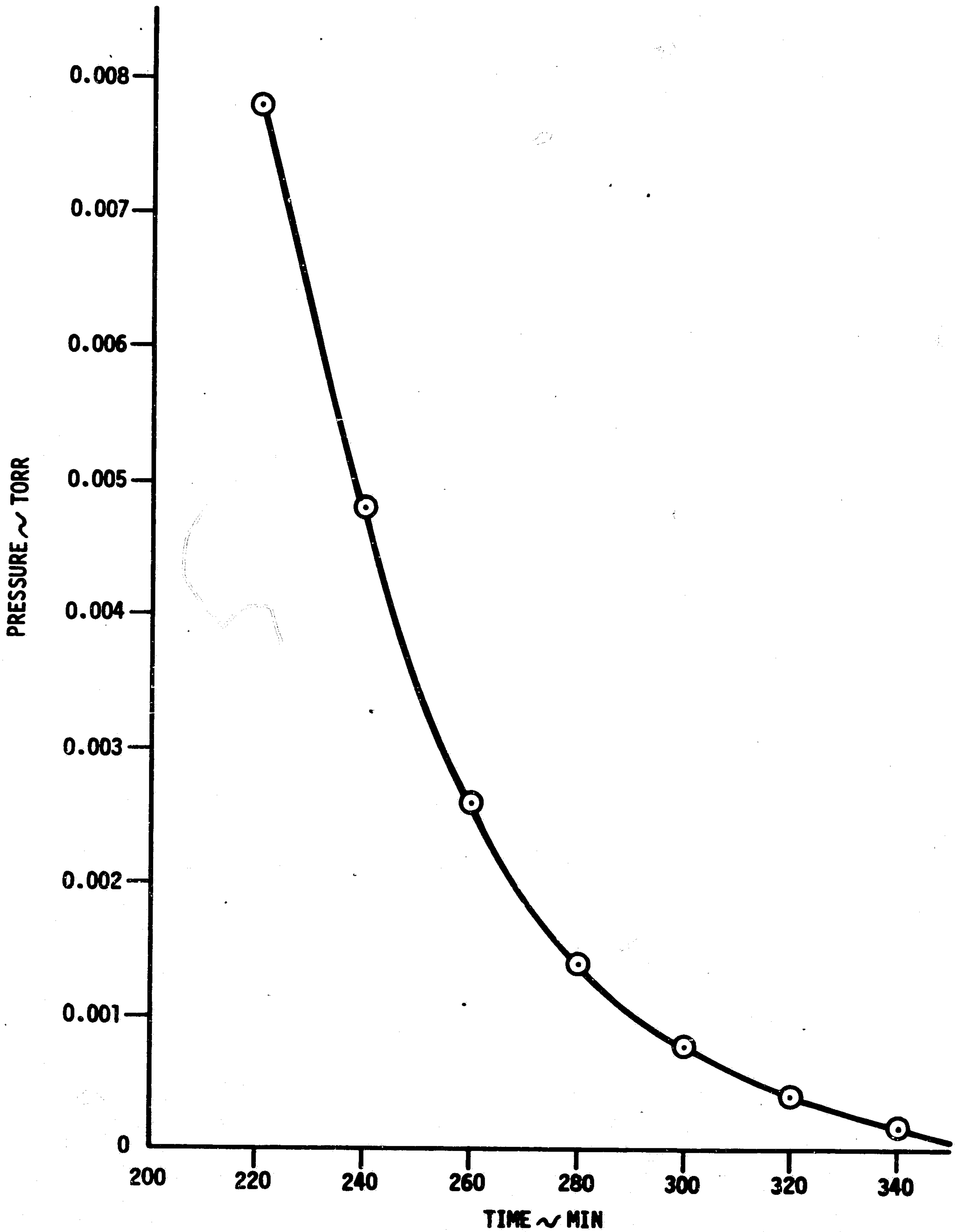


Figure 4.1-13 Mylar Insulation Venting

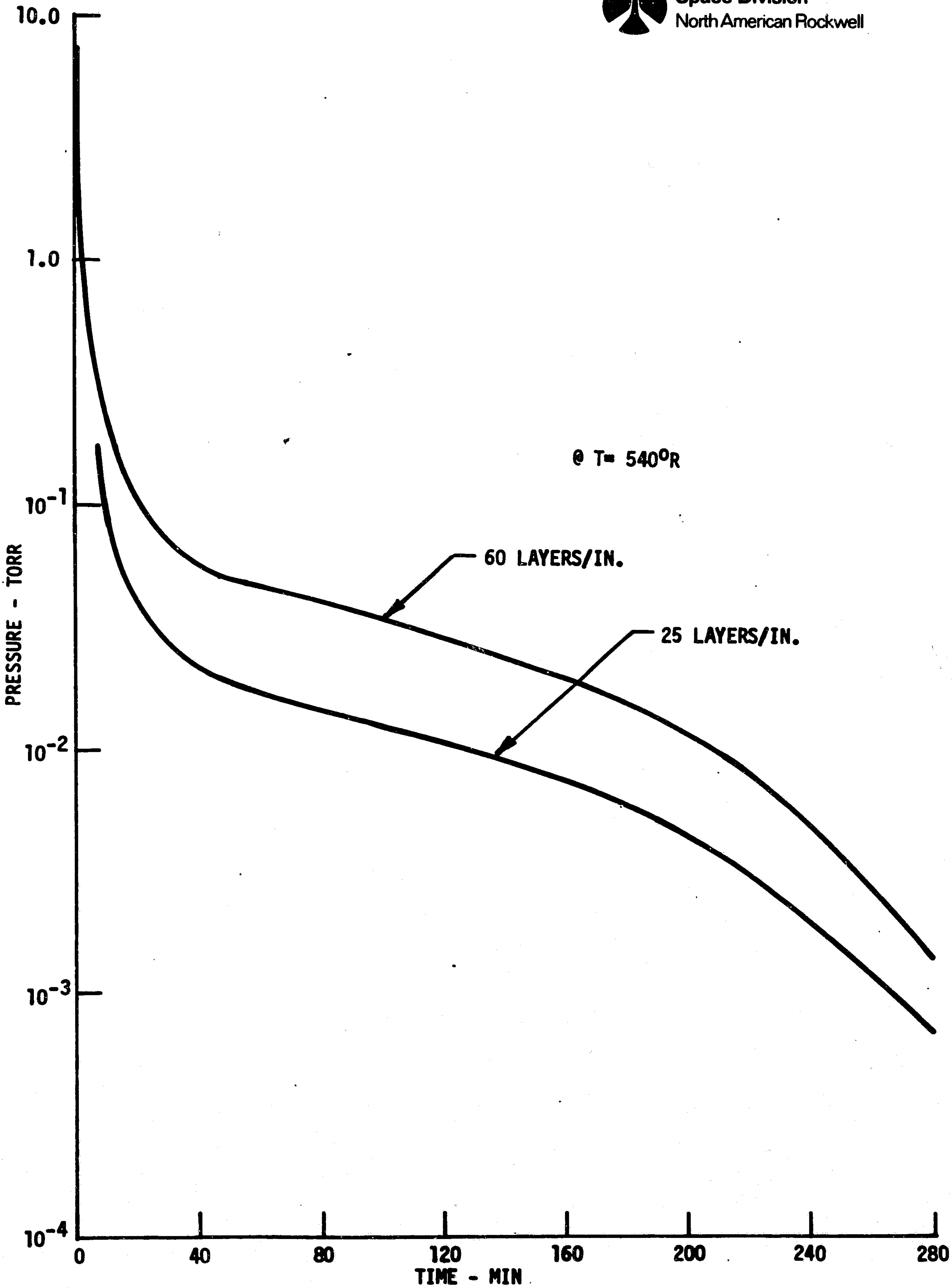


Figure 4.1-14 Aluminized Mylar Insulation Venting Layers Spacing Effect

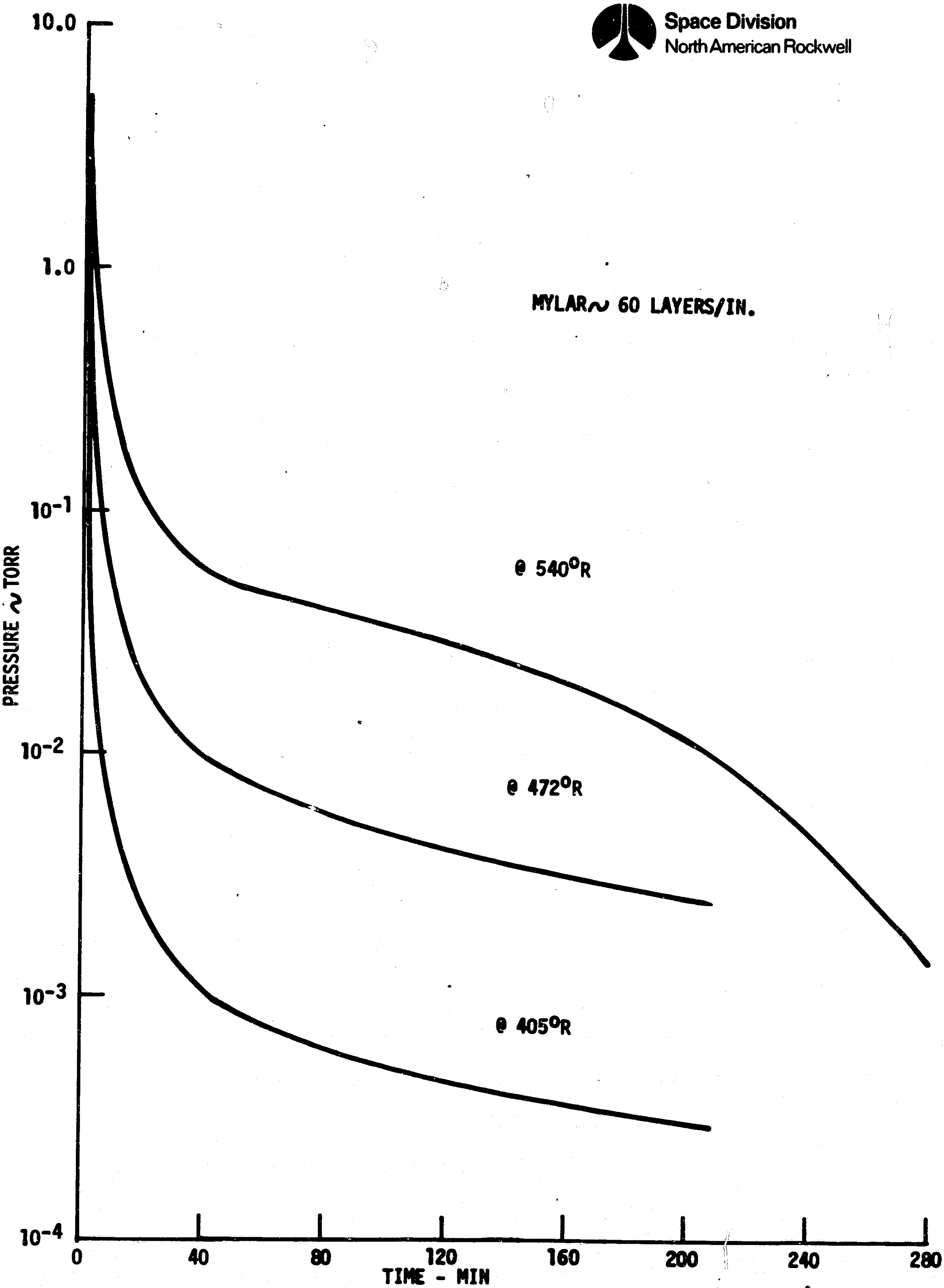


Figure 4.1-15 Aluminized Mylar Insulation Venting Temperature Effect

a temperature of 300°K and extended to 1°K by means of a low temperature thermodynamic properties program for LOX/LH_2 mixtures.

The main engine exhaust plume iso-Mach plot is shown in Figure 4.1-16 while the RCS roll and pitch engine plot is shown in Figure 4.1-17. Note that although the RCS roll and pitch engines have different thrusts (See Section 4.3.5), the non-dimensional distribution of the exhaust plume properties are the same since the roll engine is assumed to be a scaled down version of the pitch engine.

RCS Exhaust Plume Impingement Heating Rates

RCS exhaust plume impingement region centerline impingement heating rates to the Tug surface were computed by means of the following laminar and turbulent convective heating rate relationships for flow having varying external velocity and density. The equation

$$St_x = \frac{0.329\ \mu_x\ G_x^{0.435}\ P_r^{-2/3}}{\left(\int_0^x \mu_x\ G_x^{1.87}\ dx\right)^{1/2}}$$

is used for laminar flow, and the equation

$$St_x = 0.0295\ Re_x^{-0.2}\ P_r^{-0.4}$$

where

$$R_{e_x} = \int_0^x \frac{\rho v}{\mu}\ dx$$

is used for turbulent flow. In the above equations

P_r = Prandtl number

St = Stanton number

μ = Viscosity

$G = \rho v$

ρ = density

v = velocity

x = distance from boundary layer origin

R_e = Reynolds number

The heat flux $\dot{q}$ was computed from the equation

$$\dot{q} = St\ \rho v(i_r - i_w)$$

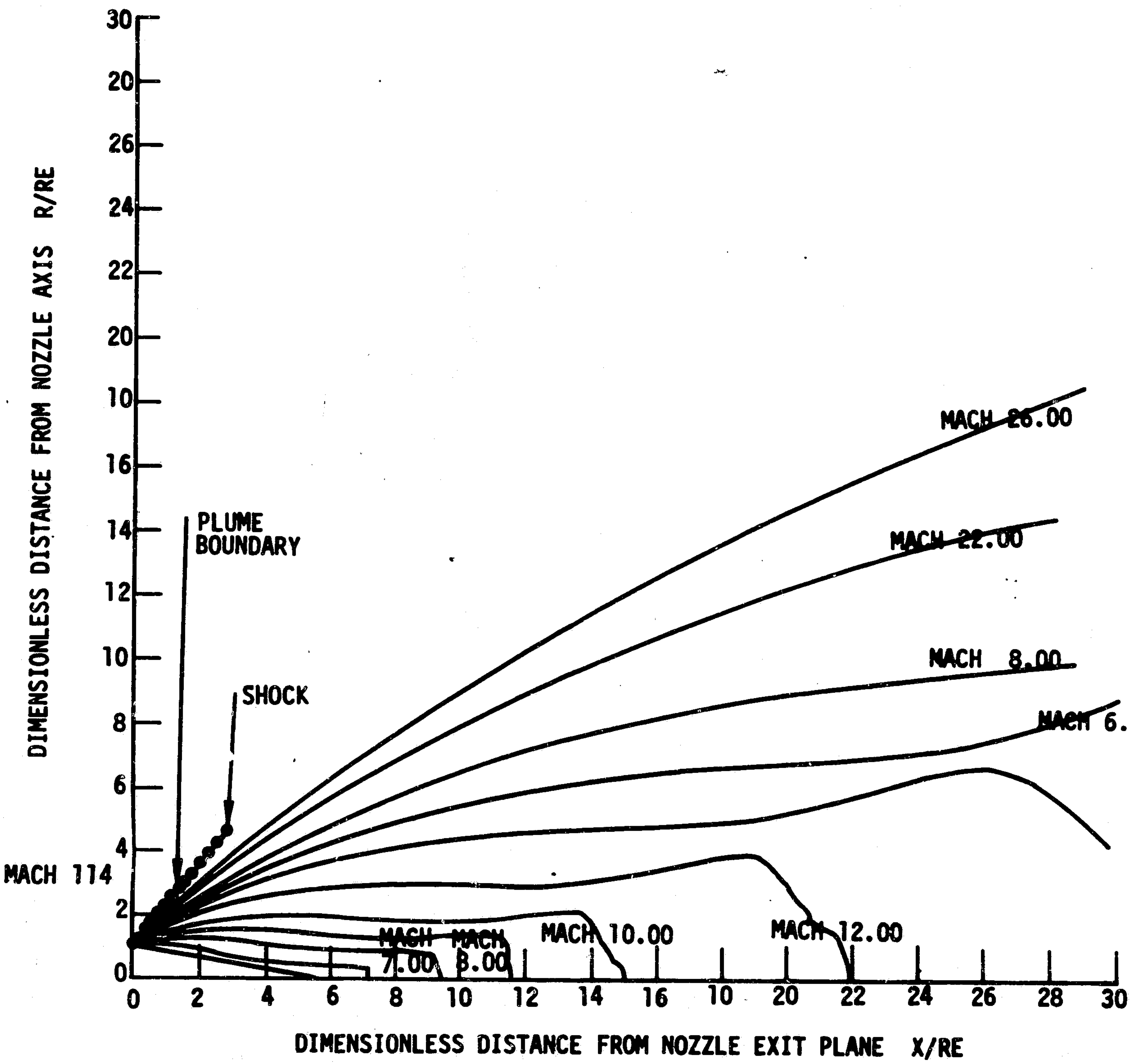


Figure 4.1-16 Tug Main Engine Isomachs MR-6.0, PC = 2020, TC = 3571 H-600K

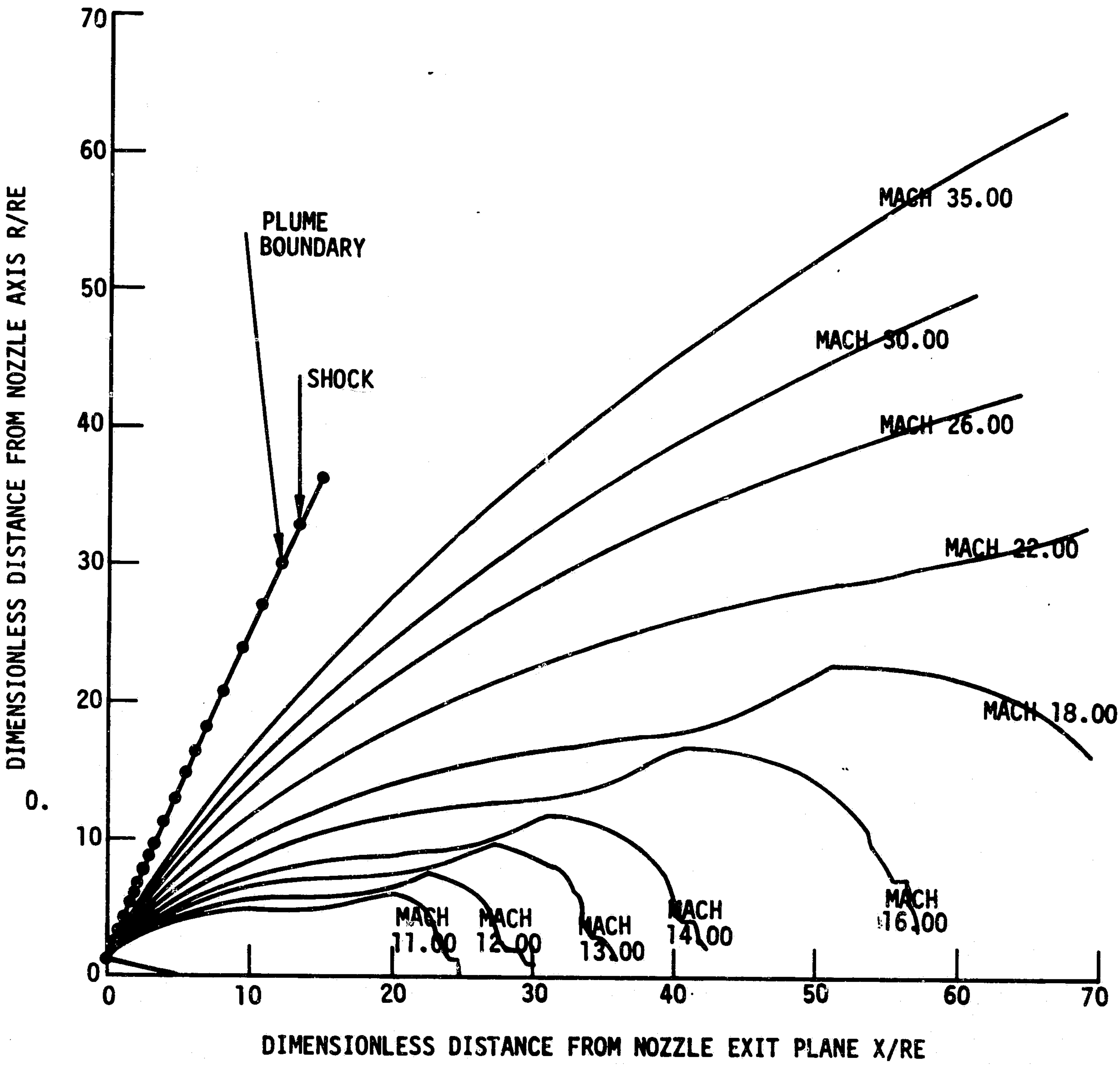


Figure 4.1-17 Tug APS Engine Isomachs MR-4.0, PC = 250, TC = 2900 H-600K

where the recovery enthalpy i_r is given by

$$i_r = i_\infty + rv^2/2\ gJ$$

and the recovery factor r is related to the Prandtl number P_r by

$r = P_r^{1/2}$ for laminar flow, and

$r = P_r^{1/3}$ for turbulent *flow.*

The impingement region flow properties were computed by means of Newtonian impact theory using the local undisturbed plume flow properties for the upstream conditions.

The impingement region centerline heating rates for the RCS pitch and roll engines are presented in Figures 4.1-18 and 4.1-19, respectively, for both the zero and 20 degrees nozzle cant configurations. It is seen that canting the nozzles 20 degrees from the impingement surface results in a reduction of approximately 60% in the peak impingement heating rates.

Figures 4.1-20 and 4.1-21 present the heating rate distribution for the RCS forward firing pitch and roll engine impingement regions, respectively. These data were estimated by determining (for any downstream station) the lateral variation of stagnation enthalpy flux normal to the surface and normalizing this distribution by the previously computed local centerline heating rate. It should be noted that the data presented in Figures 4.1-18 through 4.1-21 are applicable for a smooth surface only and appropriate protuberance factors should be applied as required.

Tug Main Engine Nozzle Heating Rates

The main engine nozzle heating rates resulting from the aft firing RCS pitch engines (zero cant) are presented in Figure 4.1-22. Since RCS plume impingement occurs near the plume boundary where the flow is highly expanded and approximately normal to the nozzle surfaces, these data were computed from the local value of the stagnation enthalpy flux multiplied by 0.6. These heating rates are most probably on the high side since the flow based on the nozzle exit plane diameter appears to be in the continuum flow regime while the 0.6 factor implies transitional flow regimes and would include the effect of engine hat bands and other surface discontinuities.

Main Engine Radiative Heating

Radiative heating to the Tug base region due to main engine exhaust plume radiation is presented in Figure 4.1-23 for the undeflected engine and for a 7 degree engine deflection case. The radiative heat flux was computed by means of a single-plume radiative heat transfer program using Hottel's approximate method of total emissivity gradient to compute radiative intensity along various lines of sight passing through the gaseous plume.

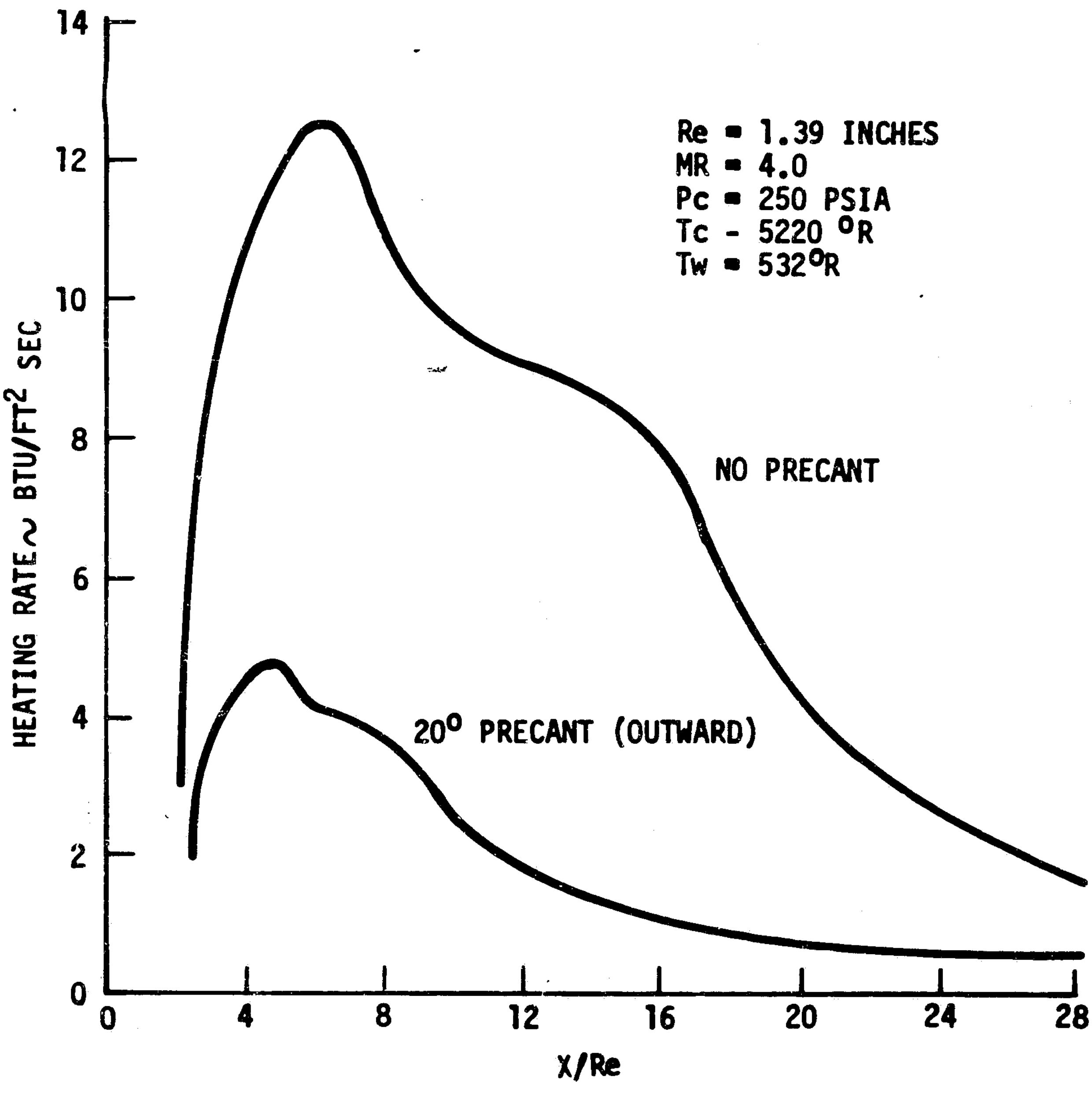


Figure 4.1-18 Space Tug APS Fwd Firing Pitch Engine Plume Impingement Heating Rates

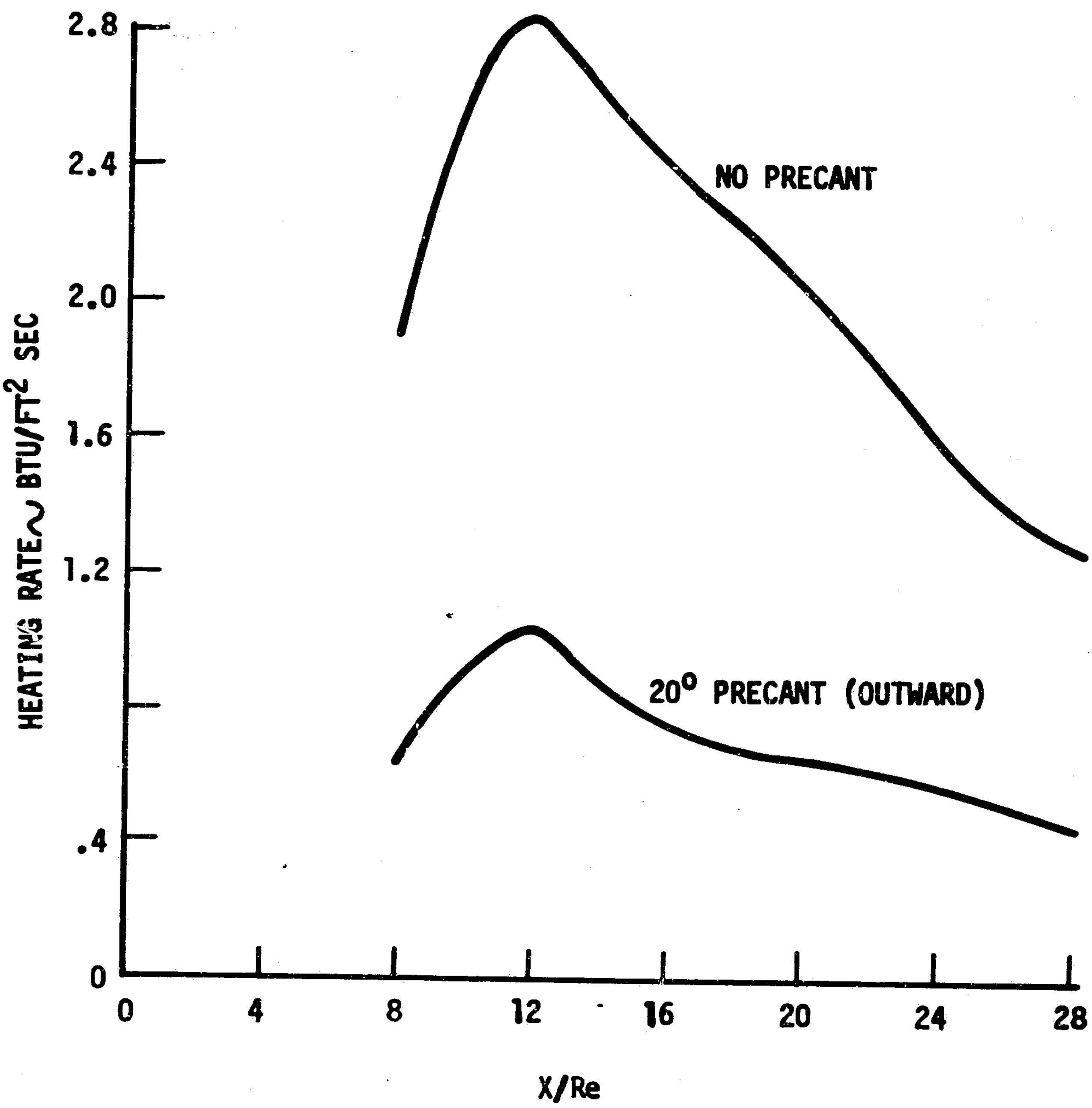


Figure 4.1-19 Space Tug APS Roll Engine Plume Impingement Heating Rates

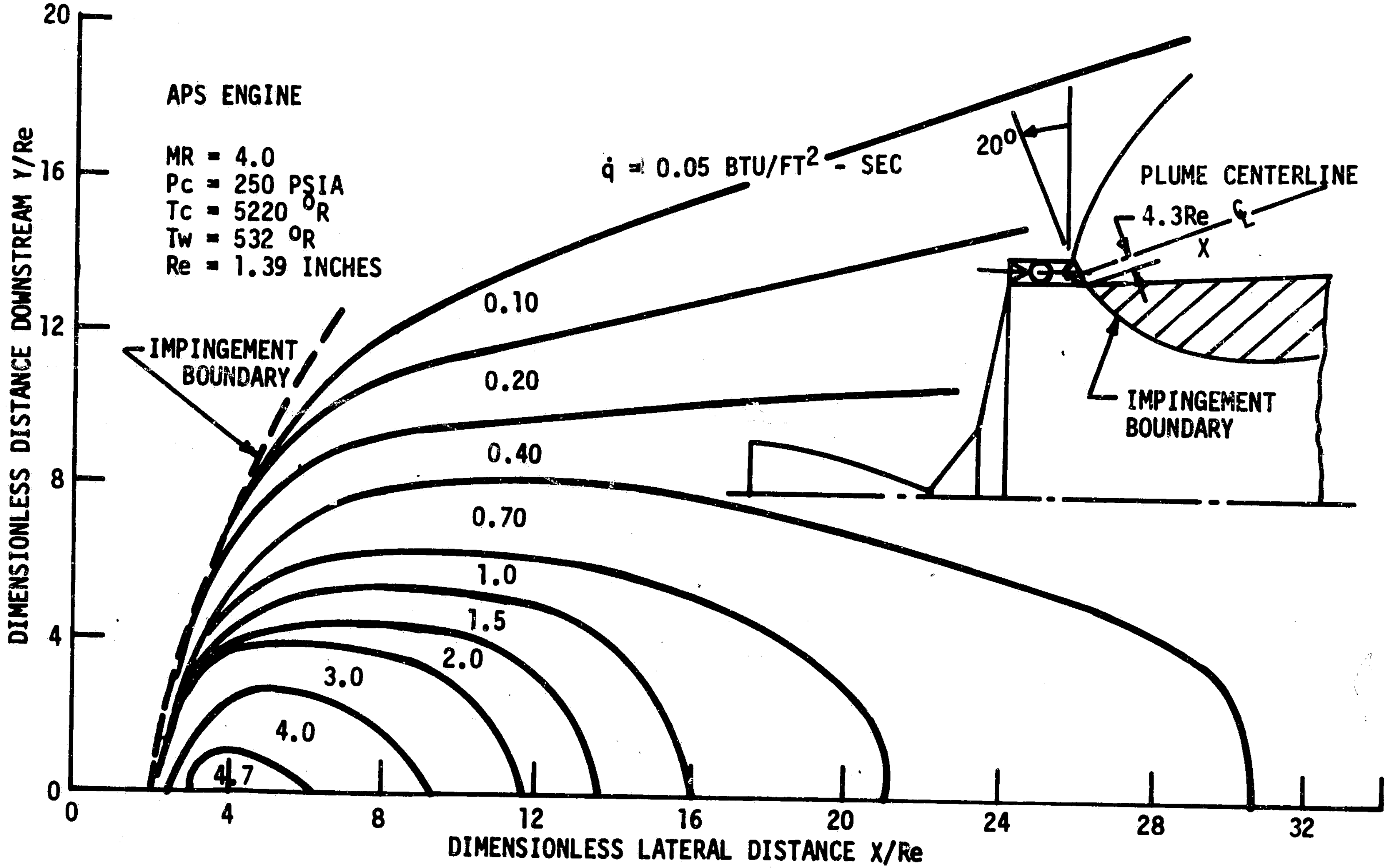


Figure 4.1-20 Space Tug Heating Rates Due to Forward Firing ACPS Pitch Engine

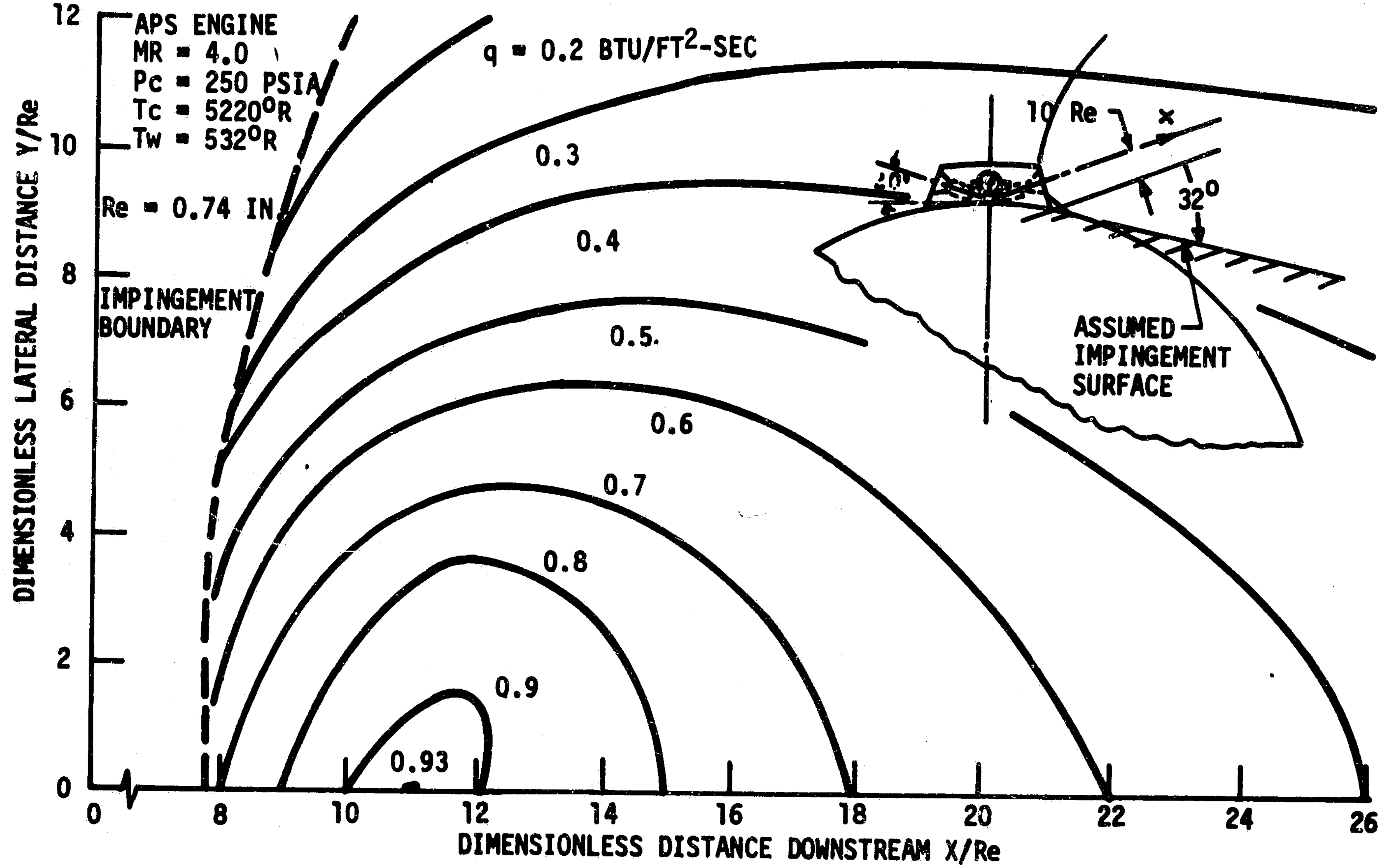


Figure 4.1-21 Space Tug Heating Rates Due to APS Roll Engine Plume Impingement

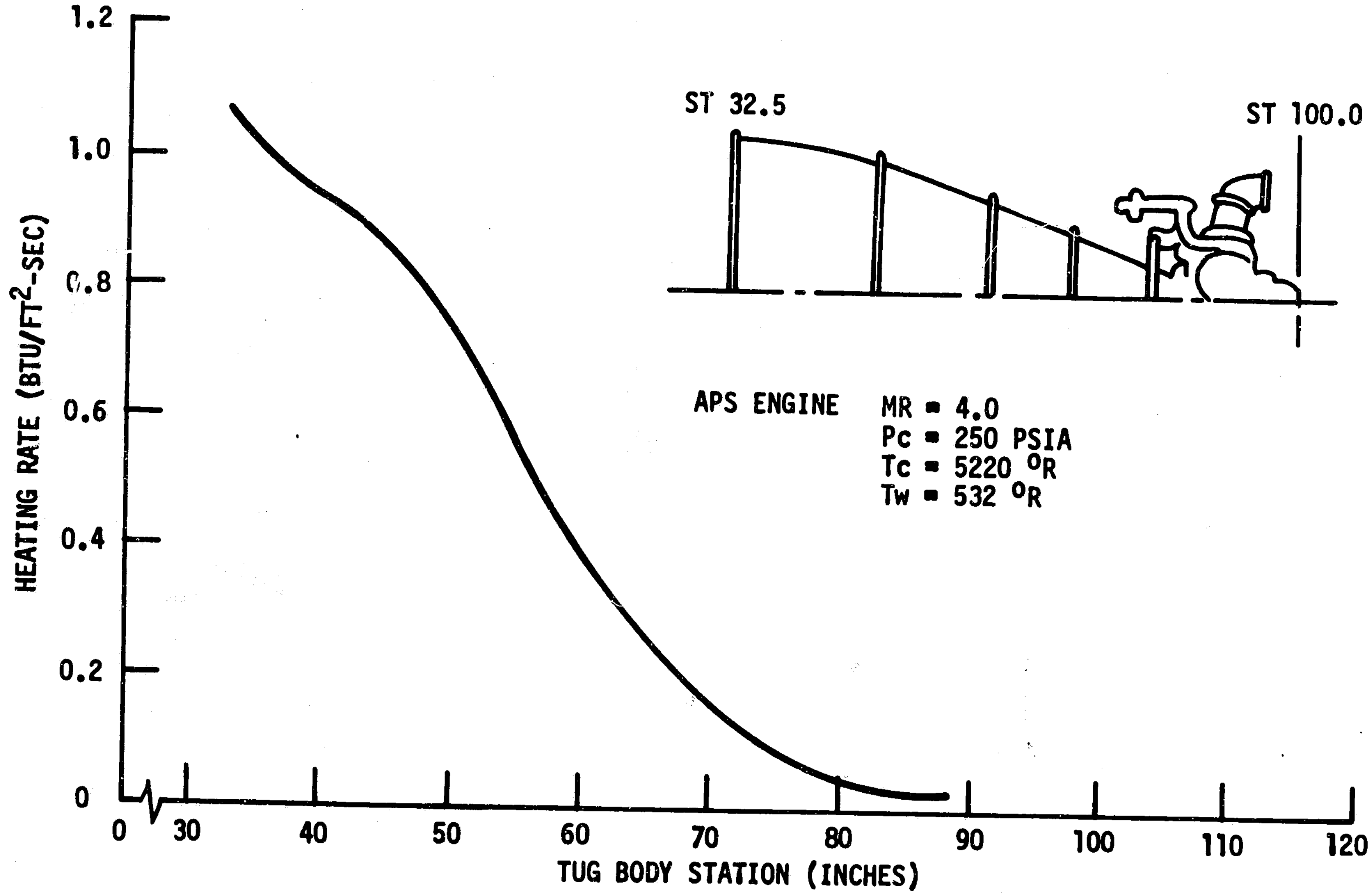


Figure 4.1-22 Space Tug Main Engine Heating Rates Due to APS Plume Impingement

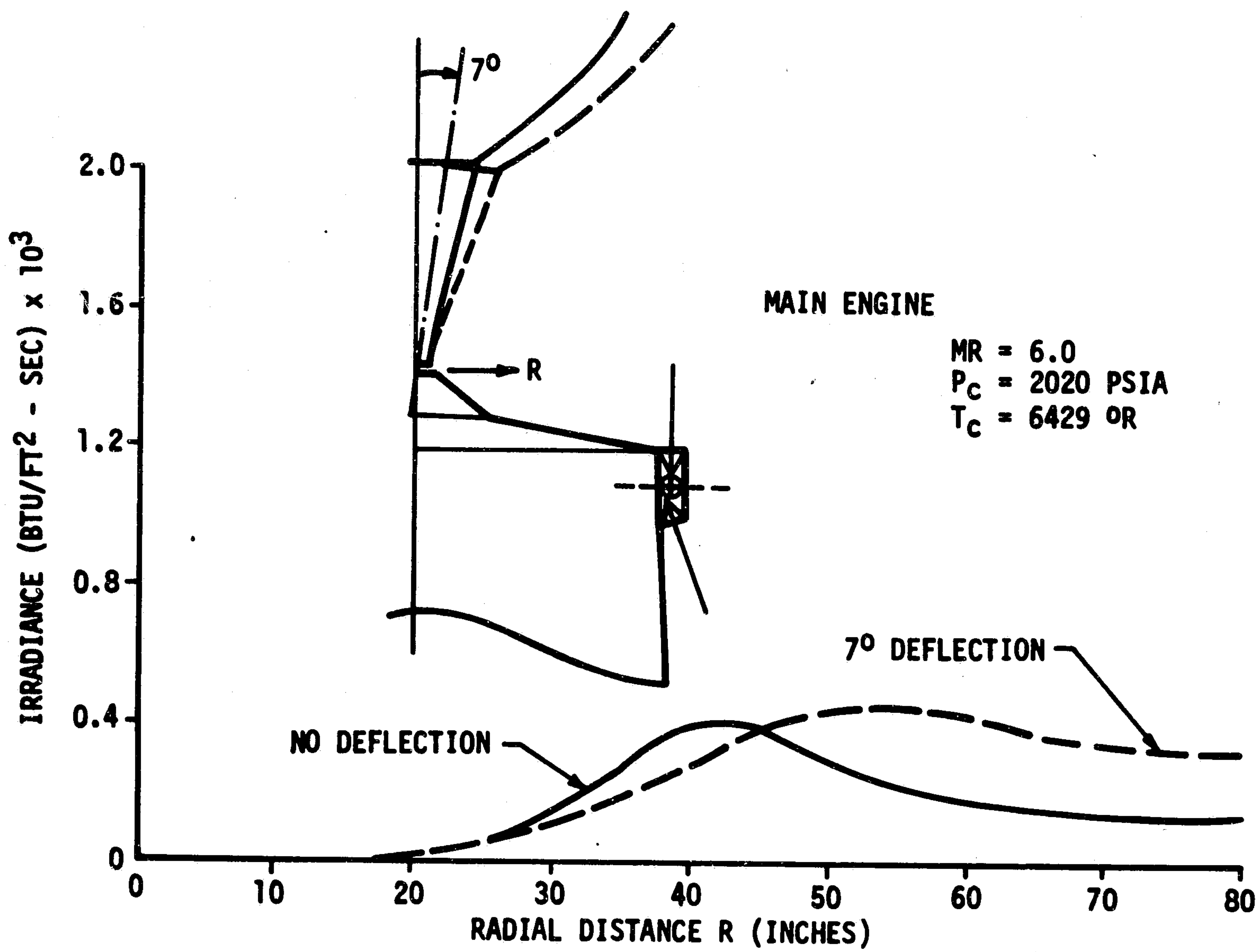


Figure 4.1-23 Space Tug Thrust Structure Heating Due to Main Engine Plume Radiation

Because of the main engine high expansion ratio, the plume static temperature and pressures are very low. Consequently, the incident radiative heat flux to the Tug base region is very low as shown in Figure 4.1-23. Also, it is seen that a 7 degree engine deflection produces a shift in the base region incident heat flux distribution without changing the peak value of approximately 4×10^{-4} BTU/Ft^2-sec.

Conclusions

It was found that RCS plume impingement produced appreciable heating rates to the Tug surface. Canting the RCS engine nozzles to 20° from the local impingement surface reduces the peak heating rates by approximately 60% to 2.85 and 0.93 BTU/Ft^2-sec for the forward firing pitch engine and the roll engine impingement regions, respectively.

The maximum impingement heating rate to the main engine is estimated to be 1.08 BTU/Ft^2-sec and occurs at the exit plane.

Radiative heating to the Tug base region from the main engine exhaust plume was found to be negligible, the peak value being approximately 4×10^{-4} BTU/Ft^2-sec.

4.2 THERMAL CONTROL AND PROTECTION SYSTEMS

4.2.1 Propellant Heat Leak Analysis

The purpose of this analysis was to determine propellant boiloff in order to select an optimum multilayer insulation (MLI) system for the LH_2 and LOX tanks. Boiloff of propellants results from heat transfer to the propellants in the areas indicated in Figure 4.2-1. These areas are the basic MLI installation including support posts (areas 1 and 4) and penetrations such as tank supports (areas 2 and 5), fill and drain lines (areas 3 and 7), and the thrust structure (area 6). The engine feed lines are designed with thermodynamic vents to cool the lines and therefore, it was assumed that there would be no heat transfer between the feed lines and tanks.

MLI thermal models with the Tug stored in the Shuttle cargo bay and with the Tug in orbit were used in this study. The models which were used in calculating heat transfer rates to the stage propellants were used with a previously developed general thermal analyzer digital computer program. The IBM 360 computer programs used for calculation of propellant heat transfer rates and transient temperatures is a compiler-type general heat transfer program which solves any transient or steady-state thermal problem whose finite difference equation can be represented by a simple electrical network. It is an n-dimensional program which includes heat transfer by conduction, radiation and convection. Since the heat transfer problem is represented by the analogy of an electrical network, the problem will consist of node points and conductors between the node points. The node points may have finite capacitance. Boundary node points have no capacitance and their value may be a constant or some function of other problem variables.

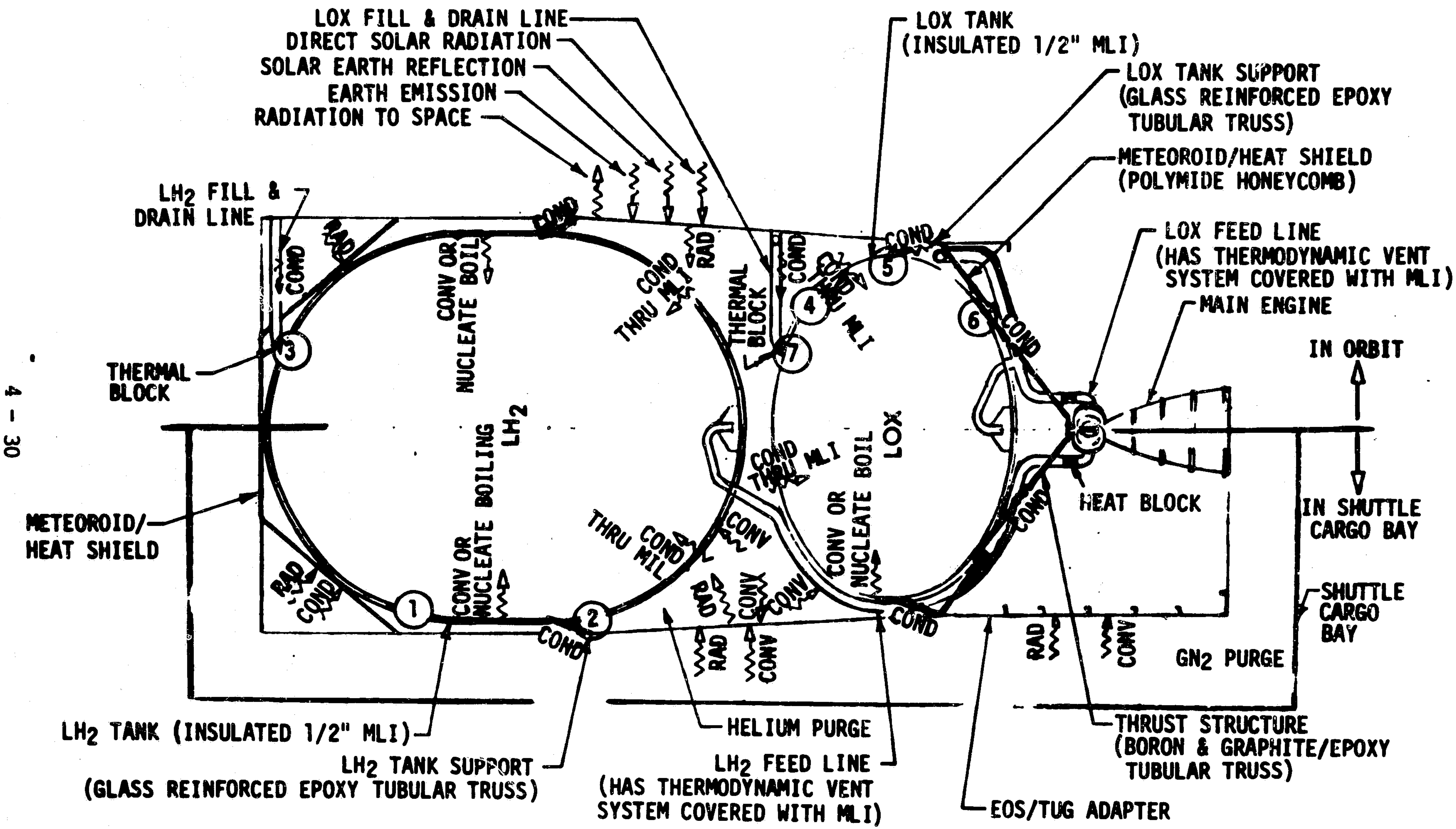


Figure 4.2-1 Modes of Heat Transfer

In the model with the Tug in the cargo bay, the Shuttle cargo bay door temperature was maintained at the temperatures shown in Table 4.2-1 and heat transfer to the Tug was by radiation and convection as indicated in Figure 4.2-1. After undocking from the Shuttle, orbital heating consisting of direct solar radiation, reflected solar radiation and earth emitted radiation was applied directly to the Tug structural wall as indicated in Figure 4.2-1. The thermal environment for the Tug while in orbit was determined utilizing a digital computer program which computes direct solar radiation, earth reflected solar radiation, and earth emitted radiation incident upon each vehicle surface at various intervals around the orbit. Heat transfer to the propellant tanks was by radiation from the Tug wall and conduction through tank supports, thrust structures and fill and drain lines.

Table 4.2-1. Environments for Propellant Boiloff Analysis

MISSION PHASE	EXTERNAL HEATING ENVIRONMENT	INSULATION INTERNAL PRESSURE
GROUND HOLD (2 MINUTES)	120°F*	760 TORR
ASCENT THRU CIRCULARIZATION (1 HOUR)	200°F*	1.67×10^{-3} TORR AT 1 HOUR
UNDOCKING AND PHASING (13 HOURS)	142 BTU/FT^2-HR	4.47×10^{-4} TORR AT 10 HOURS
TUG IGNITION THRU SHUTTLE DOCKING (126 HOURS)	142 BTU/FT^2-HR	3.63×10^{-5} TORR AT 140 HOURS
*NASA GROUND RULE		

MLI System

The study of heat leak into propellant tanks included evaluation of three variations of the embossed (spacerless) multilayer insulation (MLI) concept, all of which had a single side of the substrate with a reflective coating. The first type evaluated was NARSAM which is embossed single aluminized mylar (SAM). This MLI was included in the study because NR has done extensive research on this material and is currently under contract to the NASA to insulate 105 inch diameter test tank with NARSAM. The second type studied was single aluminized kapton (SAK). Embossed mylar has an upper temperature limit of 140°F and the ground rules established by the NASA specified maximum cargo bay environmental temperature of 200°F during both ascent and descent. The upper temperature limit of embossed kapton is expected to be well above 200°F. The third type was single goldized kapton (SGK). Gold was selected for consideration because it has a lower emittance than aluminum and because it is less sensitive to degradation by contact with moisture.

Thermal models of each of the embossed MLI variations were used with the general thermal analyzer program. Heat transfer through the MLI system was accounted for by gas conduction, MLI conduction, MLI radiation and conduction through the support posts. The gas conduction was based on equations given in "Cryogenic Systems" by Randall Barron and varied as a function of both mean temperature and MLI internal pressure. A transient pressure profile generated on the 105 inch diameter test tank program was used for the mission. Values of pressure at various times in the mission are shown in Table 4.2-1. The posts were assumed to be spaced at one for every two square feet of tank area. The MLI condition and radiation were computed from a universal equation for effective thermal conductivity developed by NR by using standard curve fitting techniques for test data. The equations used are shown below.

$$K_{eff} = K_{cond} + K_{rad}$$

$$K_{cond} = K_c \, T_m \, N^4 \times 10^{-5}$$

and

$$K_{rad} = \frac{K_r}{N \, F_e} \left(T_h^2 + T_c^2\right) \left(T_h + T_c\right) \times 10^{-10}$$

where

$$T_m = \frac{T_h + T_c}{2}, \ {}^\circ R$$

N = Insulation Layer density, shields/Inch

K_c = Conduction coefficient

K_r = Radiation coefficient

$$F_e = \frac{1}{\epsilon_1} + \frac{1}{\epsilon_2} - 1$$

ϵ = Surface or coating emissivity

These equations were programmed into the thermal models using the constants shown in Table 4.2-2.

Table 4.2-2. MLI Constants for Propellant Heat Leak Analysis

INSULATION CONCEPT	N	K_c	K_r*	ϵ_1 (Reflector)	ϵ_2 (Substrate)
SAM	60	0.3825	1.033	0.045	0.35
SAK	60	0.3825	1.033	0.045	0.40
SGK	60	0.3825	1.033	0.035	0.40

*K_r was derated 30% in the analysis due to perforations

Two concepts for purging and venting the MLI were considered in the study and are shown in Figure 4.2-2. The concept shown in Figure 4.2-2a assures a positive flow of purge gas but requires a greater system weight than the concept shown in Figure 4.2-2a which relies on natural circulation for purging. Since the propellant heat leak analysis was being conducted concurrent with the evaluation of the purge and vent concepts and a concept had not yet been selected, the heat leak thermal models were arbitrarily based on the concept shown in Figure 4.2-4b. This is conservative since this concept would have a heat leak slightly higher than that of Figure 4.2-2a because of the purge gap of concept 2a. The MLI thermal network used in the study is shown in Figure 4.2-2c.

The timeline used in the heat leak analysis was selected so as to be compatible with the timeline for the vehicle performance analysis. The total mission time was 140 hours and included a two minute ground hold, a one hour period from launch through circularization (TUG assumed inside shuttle cargo bay), a thirteen hour period for unlocking and phasing and a 126 hour period from TUG ignition through redocking with the shuttle. The TUG external heating environments are shown in Table 4.2-1. The 120°F during ground hold and the 200°F during ascent and while in the shuttle cargo bay were specified by the NASA in the study ground rules. Also, the NASA study ground rules specified that vehicle attitude control could not be constrained for thermal reasons, so the TUG solar orientation while in orbit was assumed to be broadside to the vehicle which represents the worst case environment for propellant heat leak. The solar heat rate of 142 BTU/Ft^2-Hr shown in Table 4.2-1 represents the average heat load around the TUG while broadside to the sun.

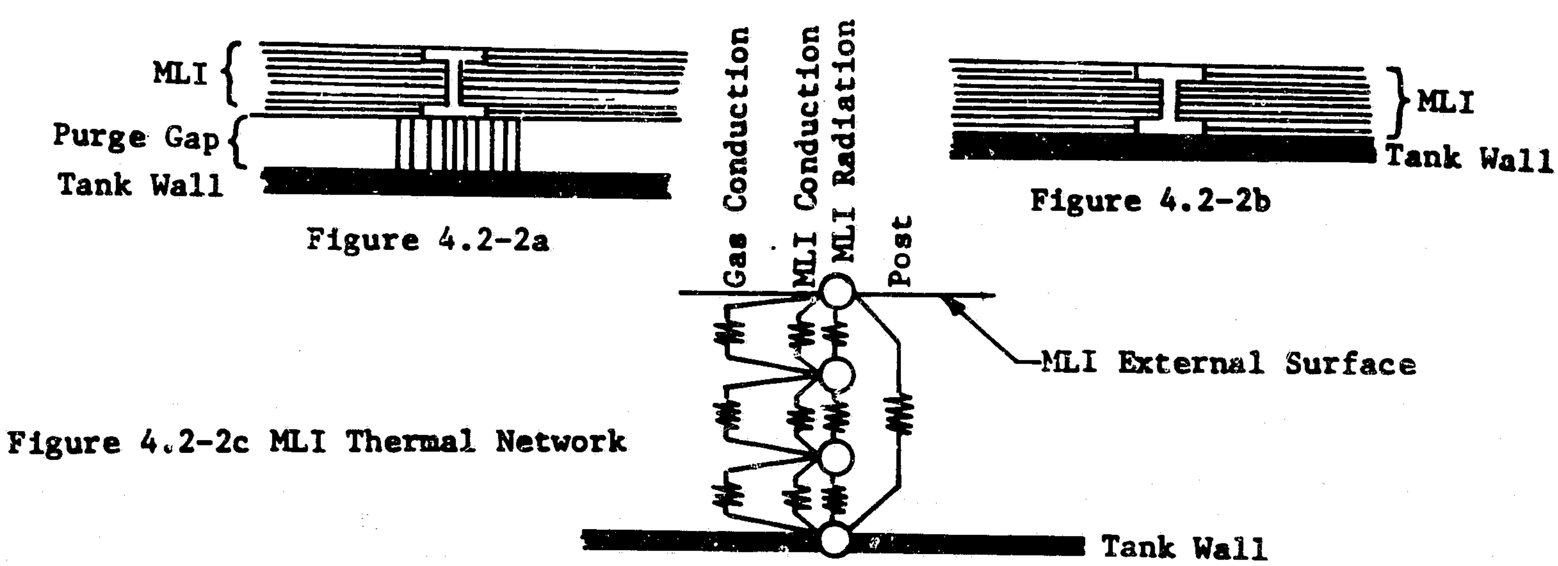


Figure 4.2-2 Purge and Vent Concepts

MLI Optimization

The results of the MLI optimization analysis are shown in Figures 4.2-3, -4, and -5 and Table 4.2-3. The baseline MLI concept and optimum thickness were selected utilizing information obtained from a study comprised of thermal analyses and vehicle performance analyses. Initial thermal analyses were conducted to determine total orbital boiloff weights (LH_2 plus LOX) for various thicknesses of three variations of the embossed (spacerless) MLI concept. The three variations were single aluminized mylar (SAM), single aluminized kapton (SAK), and single goldized kapton (SGK). The total boiloff weights calculated for the three MLI concepts are presented in Figure 4.2-3a along with the MLI weights. The boiloff weights given in Figure 4.2-3a are for the period (126 hours) between Tug ignition and redocking with the Shuttle where all the heat entering the tanks was assumed to go into boiling off the propellants. Also included in these weights are the effect of heat leaks due to tank penetrations which will be discussed later. The heat entering the tanks from the time the vent valves are closed (assumed two minutes before liftoff) to Tug ignition can be observed by the propellants without reaching saturation temperature.

Table 4.2-3. Propellant Boiloff Rates for Baseline Insulation Concept (LH_2 & LOX Tanks -0.5 Inch Single Aluminized Kapton)

MISSION PHASE	HEAT LEAK SOURCE	BOILOFF RATE (LB/HR) LH_2		BOILOFF RATE (LB/HR) LOX	
GROUND HOLD	ALL SOURCES	666 (8.3%)		478 (1.0%)	
ORBIT		LOCATION FIG. 4.2-1		LOCATION FIG. 4.2-1	
	MLI INSTALLATION	1	0.57	4	0.50
	TANK SUPPORTS	2	0.02	5	0.07
	THRUST CONE	-	-	6	0.03
	FILL & DRAIN LINE	3	0.01	7	0.01
SUBTOTAL		0.60		0.61	
CONTINGENCY TO ACCOUNT FOR INSTRUMENTATION LINE LOSSES ETC. (~5%)		0.03		0.03	
TOTAL		0.63		0.64	
TOTAL LH_2 + LOX		1.27			

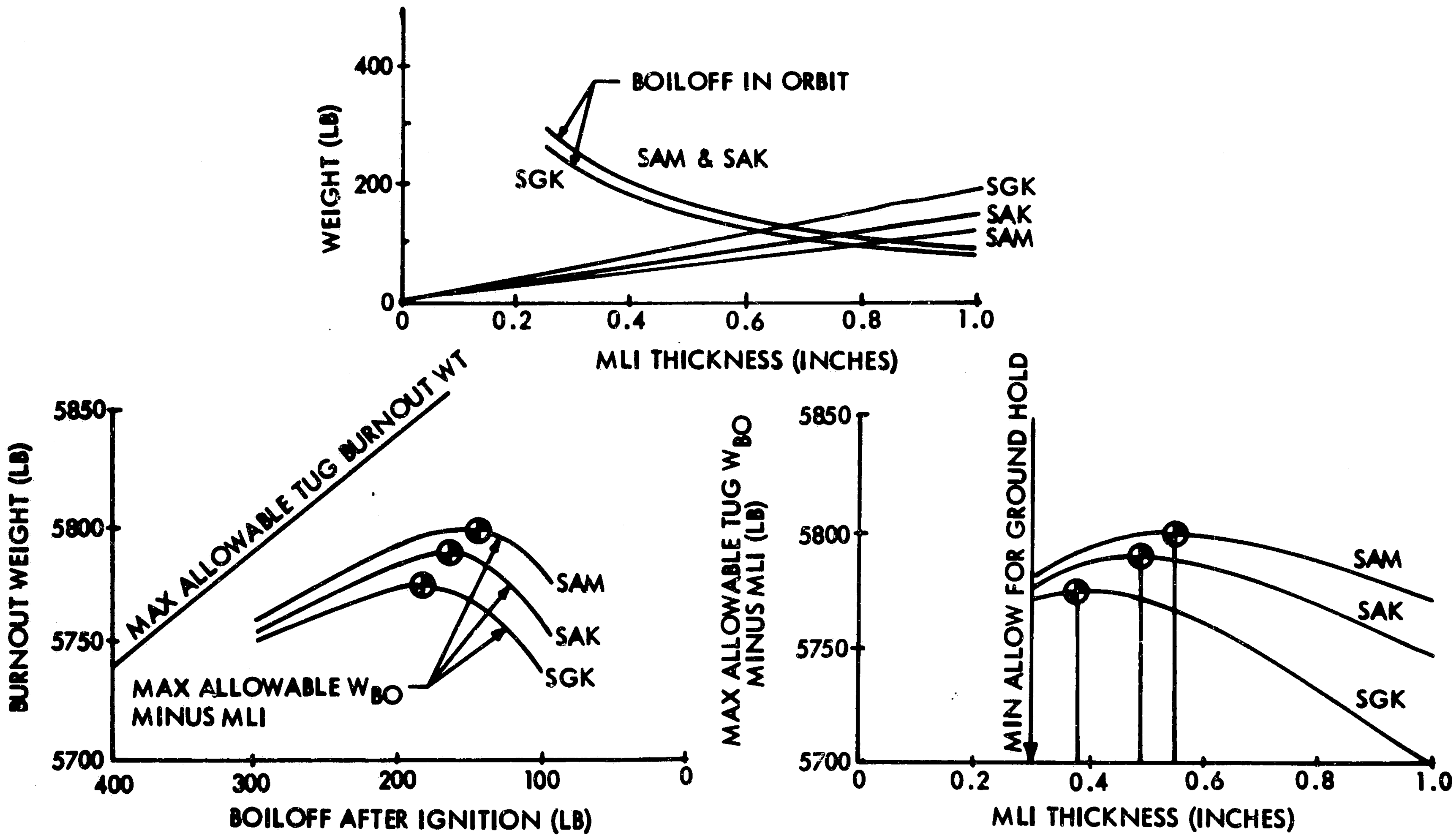


Figure 4.2-3 Insulation Concept Optimization

The information shown in Figure 4.2-3a was used in the performance analysis to generate the data shown in Figures 4.2-3b and 4.2-3c. Figure 4.2-3b a curve of maximum allowable Tug burnout weight is plotted against propellant boiloff weight after Tug ignition. The MLI weights are subtracted from the maximum allowable burnout weight to obtain the curves for each of the MLI types studies. It can be seen from the data that there is a maximum weight difference of only about 24 pounds between the three types of MLI. Therefore weight was not the prime consideration in selection of the baseline MLI system. The baseline MLI system selected was SAK and was selected because of the higher temperature capabilities of kapton and it is believed that with proper handling and conditioning, moisture will not be a problem with the aluminum coating. The data of the Figure 4.2-3b was plotted in Figure 4.2-3c versus MLI thickness in order to select the optimum thickness for SAK. Based on the curve for SAK, the optimum thickness is 0.49 inches, therefore, 0.50 inches of SAK was selected for the baseline thickness on both the LH_2 and LOX tanks.

The ground hold propellant boiloff rates are shown in Figure 4.2-4 as a function of MLI thickness. The allowable boiloff rate of 6% for LOX is satisfied for all thickness considered in this study while about 0.3 inches of MLI is required to satisfy the allowable LH_2 boiloff rate of 10%. Thus the baseline thickness of 0.5 inches satisfies all ground hold requirements.

Propellant boiloff rates for the baseline MLI concept are presented for both ground hold and orbit in Table 4.2-3. The baseline MLI concept for both LH_2 and LOX tanks is 0.5 inch of single aluminized kapton. The boiloff rates during ground hold satisfy the requirements of 10% per hour for LH_2 and 6% per hour for LOX. For the orbital mission, which includes the period between Tug ignition and redocking with the shuttle, the equivalent boiloff rates resulting from heat leak through the basic MLI installation, tank supports, fill and drain lines and thrust structure are shown. A contingency factor of approximately 5% was added to the LH_2 and LOX boiloff rates to account for heat leaks through penetrations such as instrumentation lines. This results in a total boiloff rate of 1.27 pounds per hour which amounts to 158 pounds of total boiloff for the orbital phase assuming all heat entering the tanks is used for boiloff.

Presented in Figure 4.2-5 are propellant boiloff rate histories during the entire Tug mission from ground hold through redocking with the Shuttle. This figure shows the effect of MLI internal pressure on propellant boiloff and demonstrates the need for effective venting of MLI internal gases after liftoff.

An analysis was also made to determine the effects of variation of MLI thermal properties on TUG burnout weight. For this analysis, the MLI conduction term K_c, shown in Table 4.2-2, was allowed to vary and the results are shown in Figure 4.2-6 for the baseline MLI of single aluminized kapton. Presented in Figure 4.2-6 are curves of maximum allowable TUG burnout weight minus MLI weight for MLI conduction based on K_c of Table 4.2-2 and on K_c increased by a factor of ten. The results show that although the MLI optimum

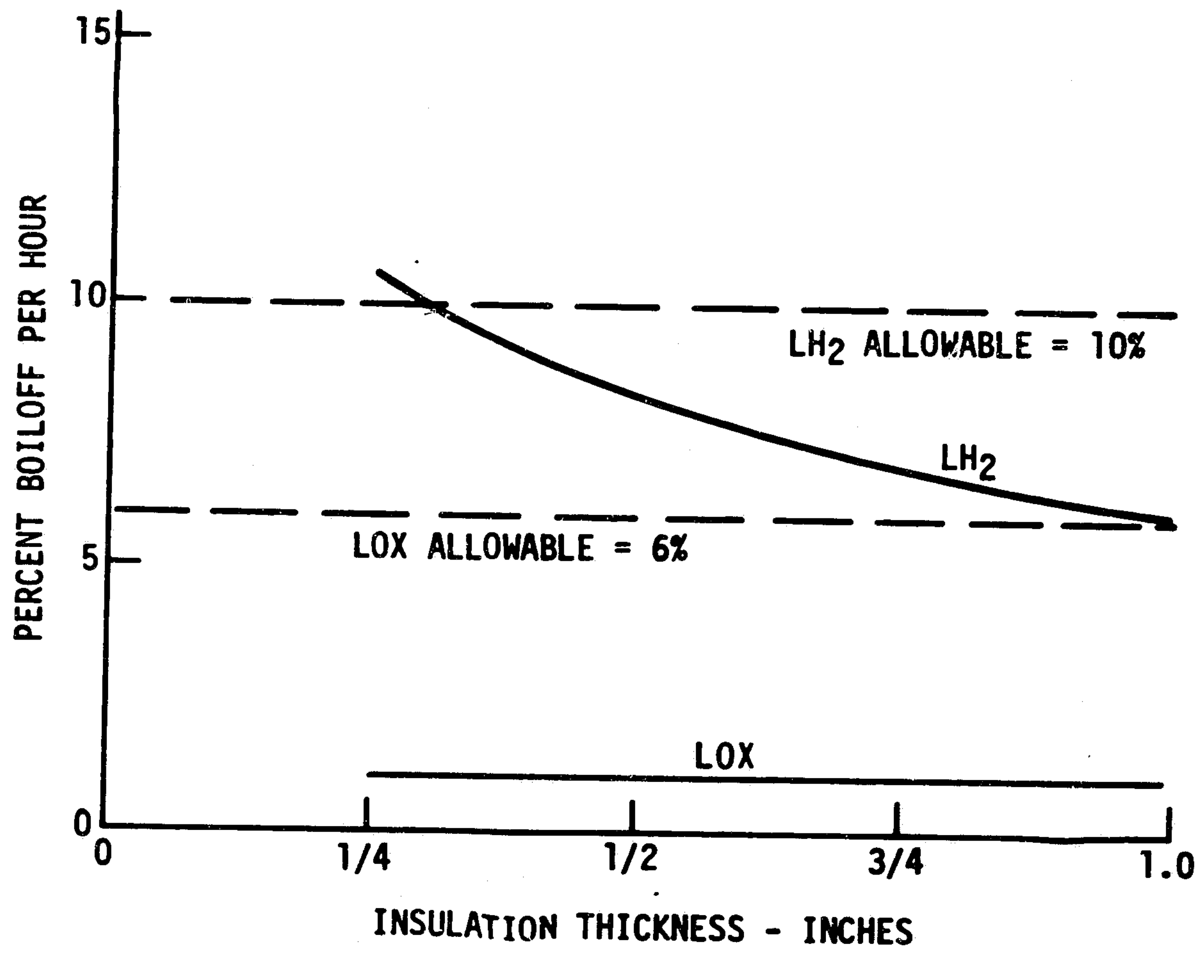


Figure 4.2-4 Ground Hold Propellant Boiloff Allowables

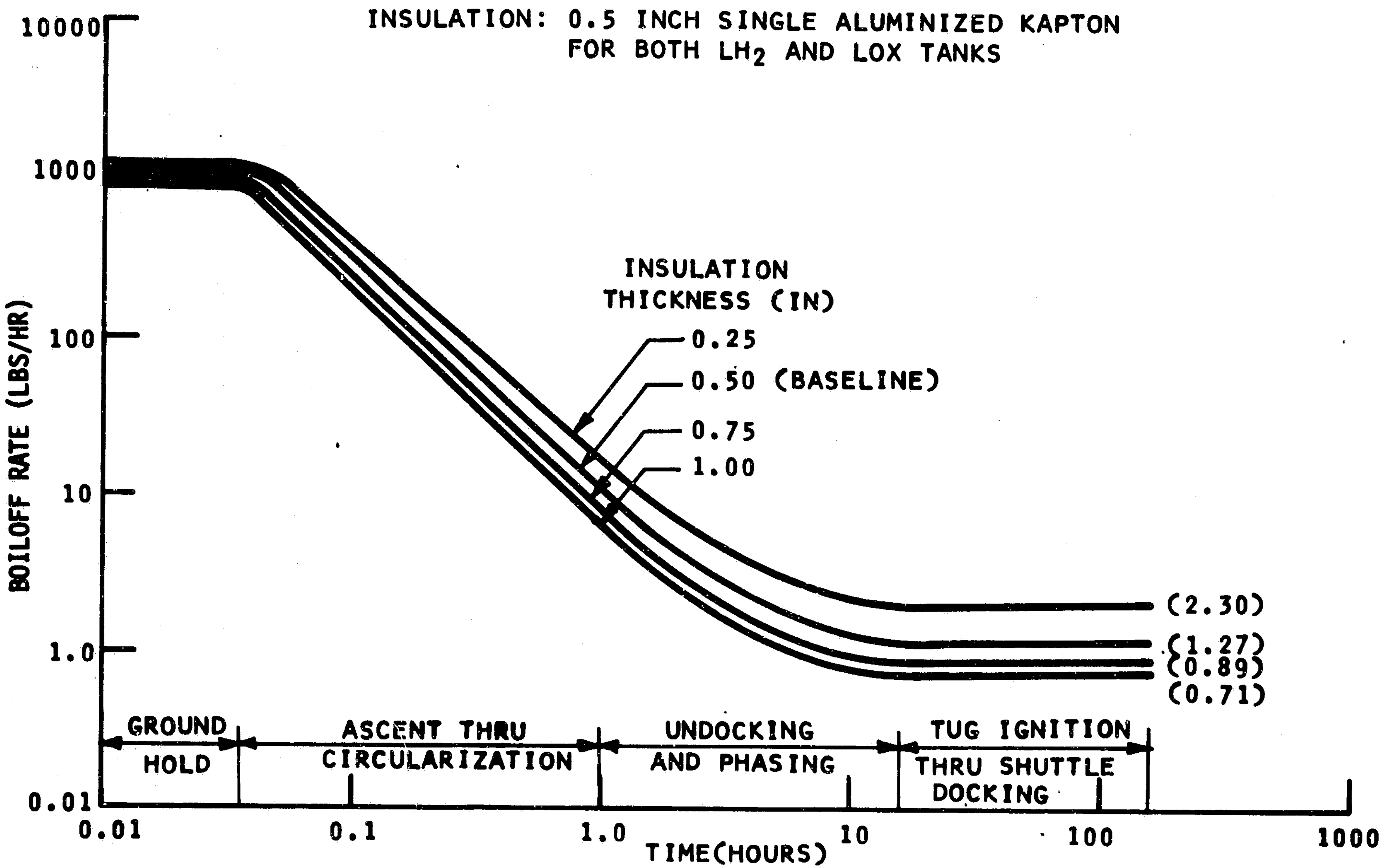


Figure 4.2-5 Propellant Boiloff Rate History

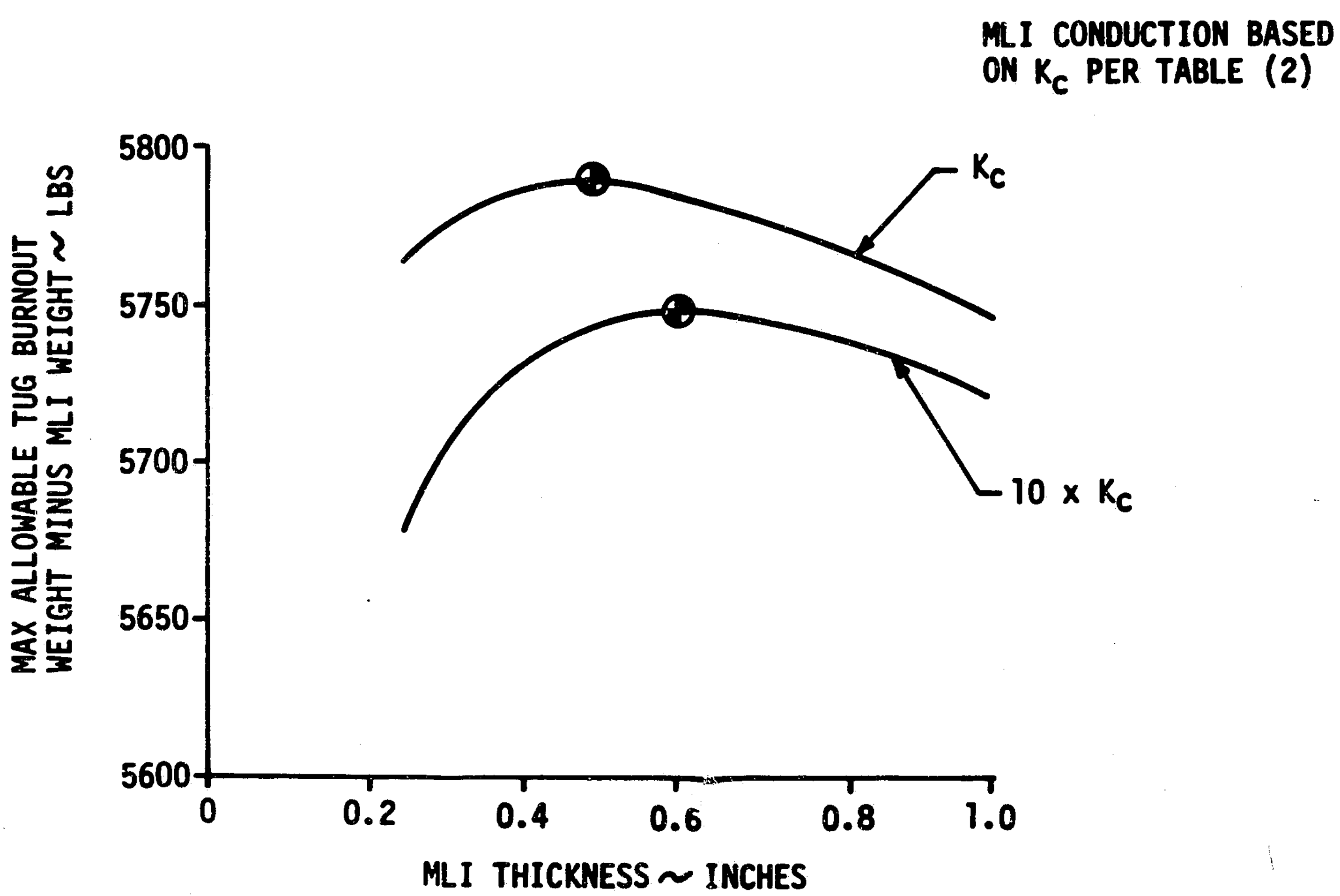


Figure 4.2-6 Effects of MLI Conduction on Tug Burnout Weight
MLI: Single Aluminized Kapton

thickness will be increased from 0.49 to 0.60 inches with higher MLI conduction, the resultant decrease in allowable TUG burnout weight is only about 42 pounds.

Conclusions

The results of the propellant heat leak analysis indicate that a baseline MLI of 0.5 inches of single aluminized kapton (SAK) is optimum for both the LOX and LH_2 tanks. The use of single aluminized mylar (SAM) would result in a slightly higher maximum allowable TUG burnout weight. However, maximum environmental temperatures of 200°F preclude the use of SAM. The maximum allowable ground hold boiloff rates of 10% per hour for LH_2 and 6% per hour for LOX are satisfied with the baseline MLI. The total orbital boiloff rate for both LH_2 and LOX was calculated to be 1.27 pounds per hour.

4.2.2 Tank Penetrations

Heat leaks through the tank penetrations must be reduced to a minimum to achieve a highly effective cryogenic tank insulation system. The major penetrations of the space Tug propellant tanks are: (1) the tank supports (which are identified in Figure 4.2-2 as Locations 2 and 5); (2) the propellant tank fill and drain lines (Locations 3 and 7); and (3) the thrust structure (Location 6). To minimize the heat leaks by conduction through these tank penetrations, low thermal conductivity materials such as fiberglass epoxy and boron epoxy combined with graphite epoxy are used. The selected composite materials provide not only a low thermal conductivity, but also a relatively high structural strength and yet they are light in weight.

Tank Supports

The LH_2 and LOX tanks of the Tug are supported from the outer shell structure by tubular truss works. Each truss works is comprised of 48 tubular trusses which are assembled in a W form around the tank circumference at Station 327 for the LH_2 tank and at Station 178 for the LOX tank. The trusses are made of reinforced fiberglass epoxy to reduce the heat leaks to the onboard propellants. The diameters of the trusses are 3/4 inch and 1 inch for the LH_2 and LOX tank, respectively. Other dimensions of the trusses are given in Figure 4.2-7. The hollow core of the tubular trusses is filled loosely with aluminized mylar sheets similar to those used for the MLI to reduce a heat leak by radiation through the hollow portion of the trusses.

The heat leak rates through the propellant tank supports are presented in Figures 4.2-7 and 4.2-8 for the space Tug in a prelaunch hold environment and in a space environment, respectively. The heat leak rates for the prelaunch hold environment are given as a function of the shuttle orbiter cargo bay ambient temperatures. The cargo bay ambient temperature during prelaunch hold will be influenced by the cargo bay structure temperature, condition of GN_2 used for the cargo bay purge, and condition of helium gas used for the space Tug purge. A temperature of 120°F is specified as the maximum cargo bay structure temperature for the study in the NASA directed study plan. For the 120°F cargo bay structure, the cargo bay GN_2 ambient

NOTE:

1. The propellant tanks are insulated with a 1/2 inch of NARSAM multi-layer insulation.
2. During prelaunch hold, the shuttle orbiter cargo bay is purged with GN_2, and the space between the tug outer shell and the propellant tanks is purged with helium gas.
3. Predominant heat transfer mode of the purged areas is assumed natural convection.

	L	ℓ	DIA	t	NUMBER OF TRUSSES
LH_2 TANK SUPPORT	21	12	3/4	.016	48
LOX TANK SUPPORT	21	12	1	.024	48

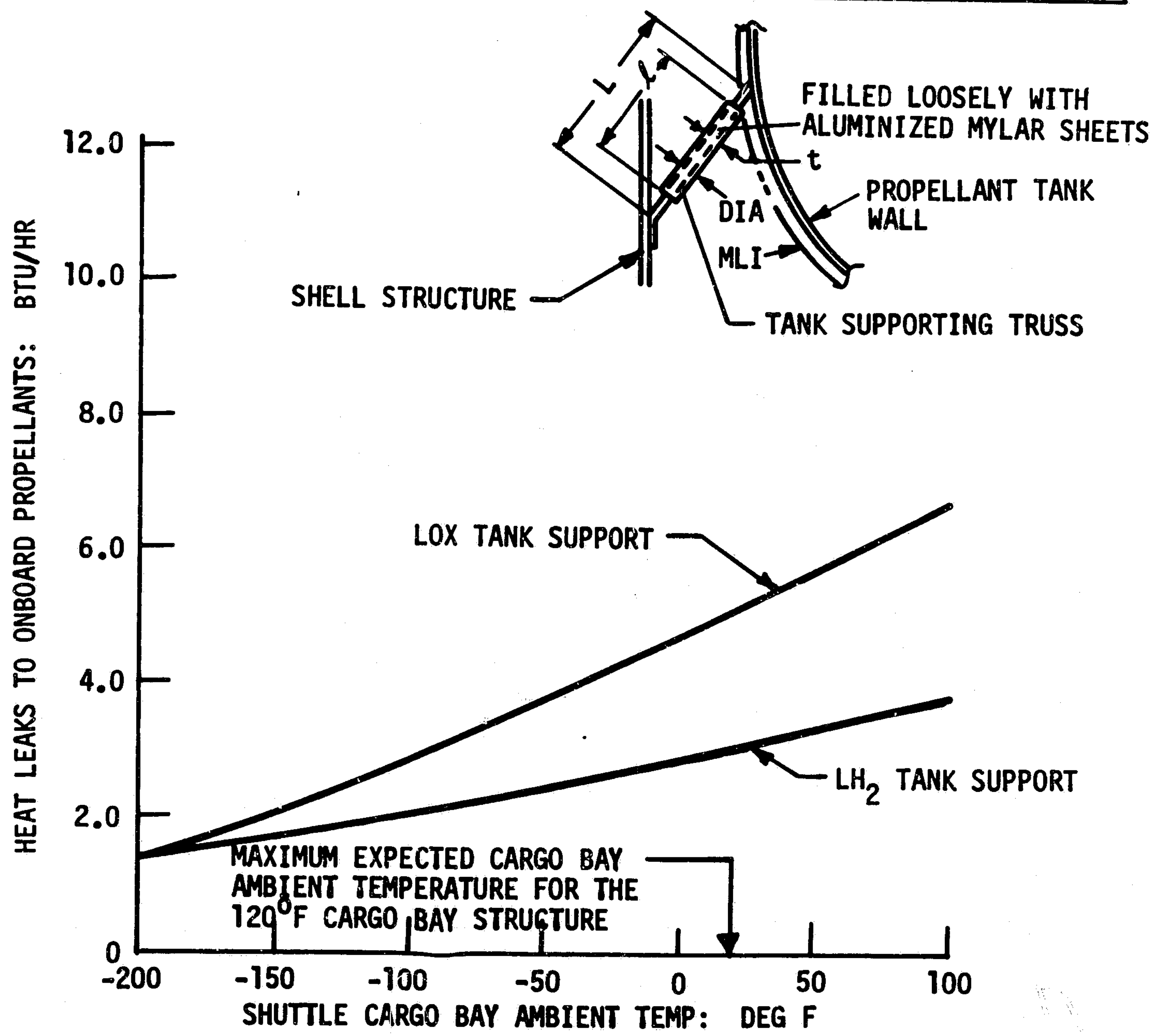


Figure 4.2-7 Heat Leaks to the Onboard Propellants Through the Tank Supports During Prelaunch Hold

NOTE:
1. THE PROPELLANT TANKS ARE INSULATED WITH A 1/2 INCH OF NARSAM MULTI-LAYER INSULATION
2. THE SHELL STRUCTURE TEMPERATURE IS ASSUMED UNIFORM AROUND THE CIRCUMFERENCE

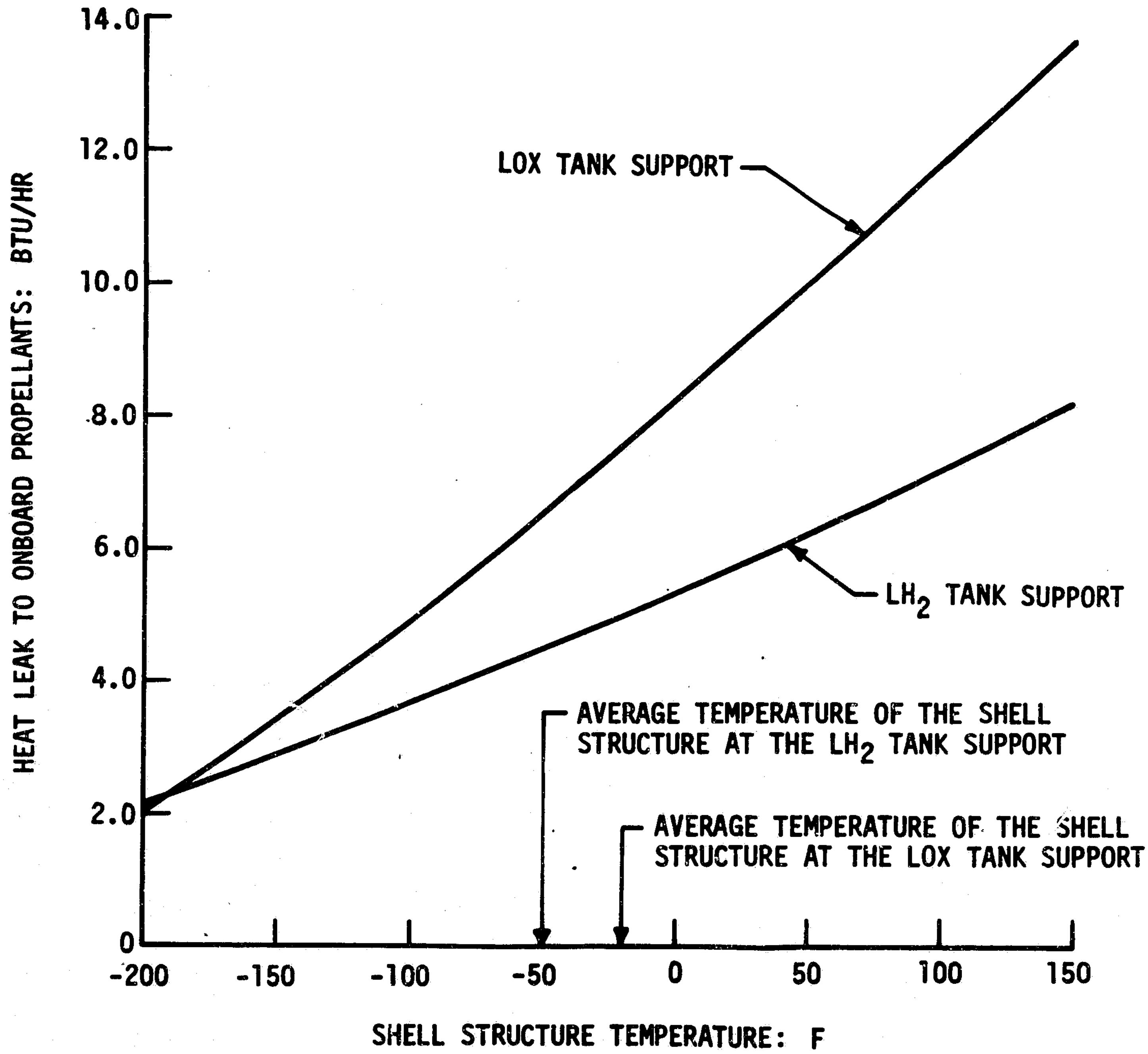


Figure 4.2-8 Heat Leaks to the Onboard Propellants Through the Tank Supports in Space Flights

temperature of about 20°F can be expected. Figure 4.2-7 shows that the heat leak rates through the LH_2 and LOX tank supports are approximately 3 BTU/Hr and 5 BTU/Hr., respectively for the 20°F cargo bay ambient. The higher heat leak of the LOX tank support is attributed to a large cross sectional area of the trusses. It is more than two times greater than that of the LH_2 tank supporting trusses.

The space flight heat leaks are given in Figure 4.2-8 as a function of the shell structure temperatures. The shell structure temperature varies with solar angle. It also varies around the periphery of the shell structure when flying broadside to the sun. The average temperature of the shell structure at the areas where the LH_2 and LOX tank supports are attached are about -50°F and -20°F, respectively. For these shell structure temperatures, the calculated heat leaks are 4.5 BTU/Hr for the LH_2 tank support and 7.5 BTU/Hr for the LOX tank support, referring to Figure 4.2-8.

Thrust Structure

The thrust structure consists of 12 tubular struts assembled in a conical shape on the LOX tank aft bulkhead as shown in Figure 4.2-9. These tubular struts are 1-1/2 inches in diameter and present heat leak paths to the onboard LOX. To minimize the heat leak, these struts are made of 2 layers of 5.25 mil boron epoxy laid in a longitudinal direction, 1 layer of a 5 mil graphite epoxy wound in a hoop direction on the inside of the boron layers and another layer of the same graphite epoxy wound in a hoop direction on the outside of the boron layers. To link these struts, a stabilizing cone is used. The stablizing cone is made of fiberglass epoxy.

The prelaunch hold heat leak rates of the thrust structure are presented in Figure 4.2-9 as a function of the cargo bay ambient temperature. For the maximum expected cargo bay ambient temperature of 20°F, the heat leak rate is 1.6 Btu/Hr. The heat leak rate during flight in space is 2.75 Btu/Hr when the thrust structure is exposed to a solar heating and no base region heating. The solar heating rate of 142 Btu/Hr-Ft^2 was assumed.

Propellant Tank Vent, Fill and Drain Lines

The fill and drain lines enter the bottoms of the LH_2 and LOX tanks while in the inverted position as shown in Figure 4.2-10. Also, the vent lines for orbit operations tap into the fill and drain lines so that there is only one tank penetration for this combination of lines. The ground vent lines tap off the engine feed lines and are routed to the same set of vent valves as the orbit vent valves.

Ground hold heat leaks were calculated for the LH_2 and LOX flight vent lines and fill and drain lines. The flight vent lines will be filled with propellants up to the vent valves while on the ground and the fill and drain lines will be filled only to the shutoff valves. In determining the heat leaks, it was assumed that the fill and drain lines are purged with helium prior to liftoff. Convective currents will be set up in the helium gas in the fill and drain lines, providing high heat leak paths. In space flight, the

NOTE:

1. THE LOX TANK IS INSULATED WITH A 1/2 INCH OF SAK MULTI-LAYER INSULATION.
2. DURING PRELAUNCH HOLD, THE SHUTTLE ORBITER CARGO BAY IS PURGED WITH GN_2, AND THE SPACE BETWEEN THE TUG OUTER SHELL AND THE PROPELLANT TANKS IS PURGED WITH HELIUM.
3. PREDOMINANT HEAT TRANSFER MODE OF THE PURGED AREAS IS ASSUMED NATURAL CONVECTION.

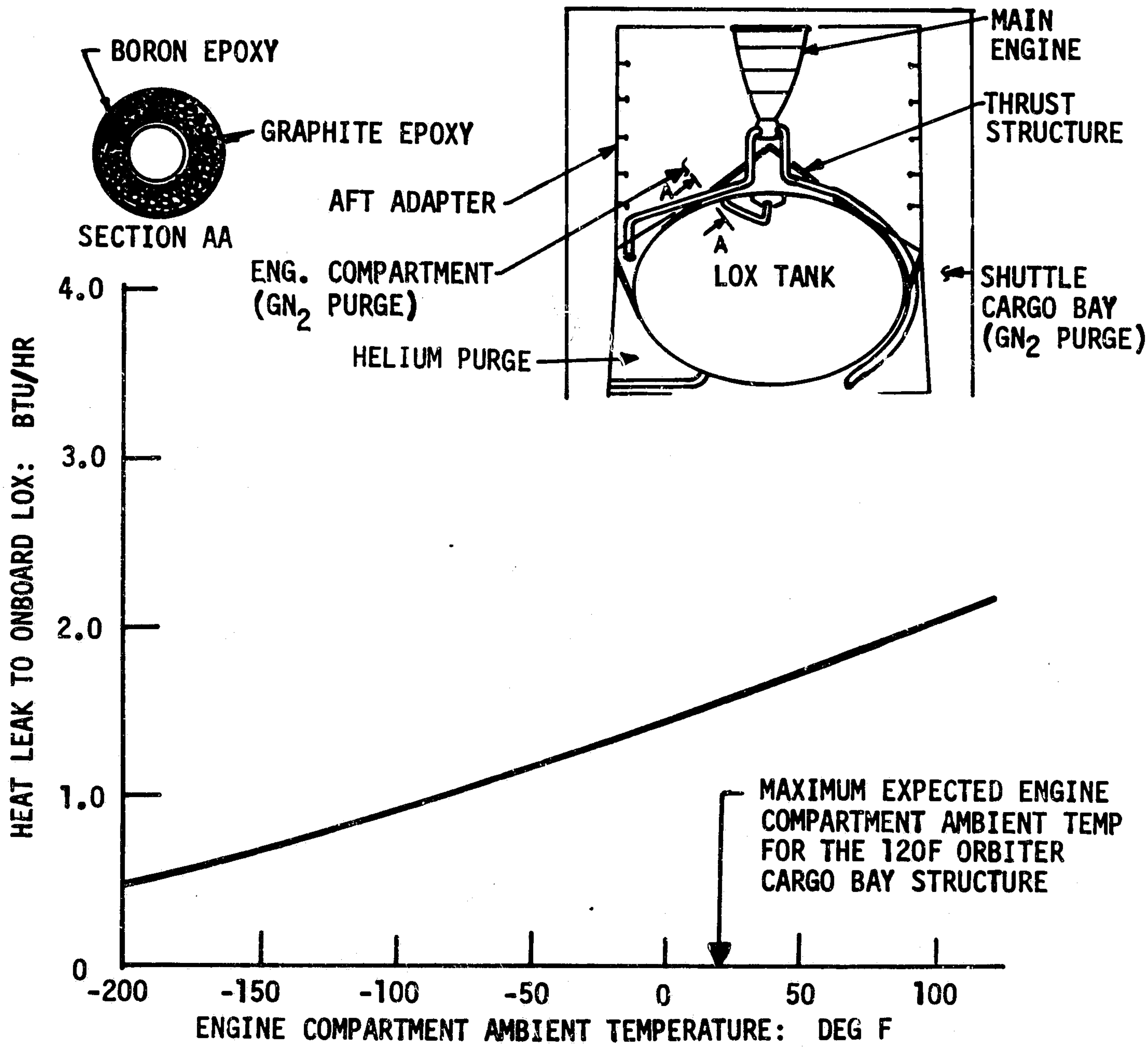


Figure 4.2-9 Heat Leak to the Onboard LOX thru the Thrust Structure During Prelaunch Hold

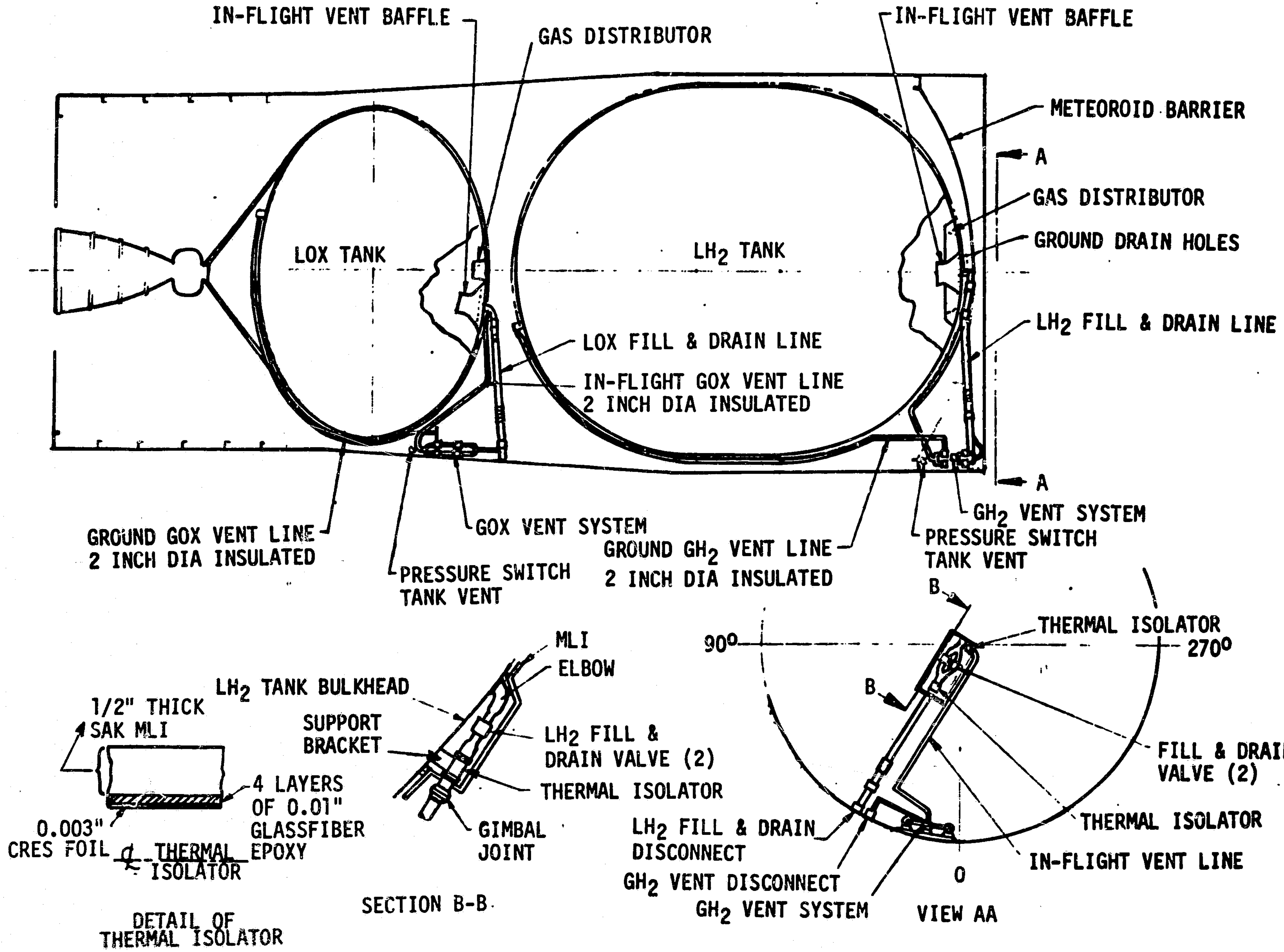


Figure 4.2-10 Space Tug, LH_2 and LOX Tank Fill & Drain and Vent Systems

helium will be vented to space, therefore eliminating heat leak by the helium convective current. About 15 inches of the LH_2 tank flight vent line is routed outside of the Tug purge bag and is subjected to warm cargo bay GN_2 purge. This warm cargo bay GN_2 contributes to the prelaunch-hold heat leaks into the LH_2 tank vent line. To reduce the heat conducting to the onboard propellants through the tank fill and drain systems, thermal isolators are incorporated in the lines immediately outboard of the shutoff valves. The thermal isolators are made of 3 mil CRES foil reinforced with 4 layers of 0.01 inch glassfiber epoxy as shown in Figure 4.2-10. It is insulated with 1/2 inches of MLI. The length of the thermal isolator was determined to be 6 inches based on the effectiveness of the thermal isolator in reducing the heat leaks in space flight and the problem associated with an installation of long ducts of this type. The isolators for both the flight vent lines and fill and drain lines were designed for orbit but the isolator for the fill and drain line also assists in minimizing ground heat leak. An insulation of 0.5 inch single aluminized kapton (SAK) was assumed for the entire length of the flight vent line. The fill and drain line was assumed insulated with 0.5 inches of SAK for ten inches outboard of the thermal isolators. The area in the vicinity of the actual tank penetration is covered by an insulated box as shown in Figure 4.2-10. The use of an insulated box reduces the complexity associated with insulating individual lines. This box is also insulated with 0.5 inches of SAK. The flight vent line and fill and drain line total ground boiloff rates for LH_2 and LOX were calculated to be 7 pounds per hour and 6 pounds per hour, respectively. These ground boiloff rates have been incorporated into the total ground boiloff rates shown in Table 4.2-3.

The heat leak rates through the LH_2 and LOX tank fill and drain lines in space flight are plotted on Figures 4.2-11 and 4.2-12, respectively, as functions of the thermal isolator length and the extension of 1/2 inch SAK coverage on the lines. The heat leaks presented in Figures 4.2-11 and 4.2-12 are based on a MLI length of 10 inches (Y) beyond the thermal isolator. The effects of the MLI length on the heat leaks are presented in Figure 4.2-11 and Figure 4.2-12. They show insignificant difference in the heat leaks with extended MLI coverages. Using the same basic analysis approach for the flight vent lines results in the vent and fill and drain lines contributing approximately 0.01 pounds per hour of boiloff for both the LH_2 and LOX tanks. These boiloff rates have also been included in the total space flight boiloff rates shown in Table 4.2-3.

The propellant boiloff while on the ground and inverted in the shuttle cargo bay will be vented overboard through ground vent lines which tap off the engine feed lines. When the vehicle is in an upright position these ground vent lines will fill up with propellant. An analysis was conducted to determine the heat leak and resultant propellant boiloff between TUG ignition and redocking with the shuttle. An insulation of 0.5 inches of single aluminized kapton was assumed for both the LH_2 and LOX vent lines. Since the vehicle orientation cannot be constrained for thermal reasons, the vehicle was assumed broadside to the sun and not rotating. The temperature of the shell with a low solar absorbtivity white coating would be in the 80 to 120°F range. Based on these environments, the heat leak rate for the LH_2 vent line is 7.0 Btu/Hr and for the LOX vent line 5.0 Btu/Hr. The corresponding boiloff

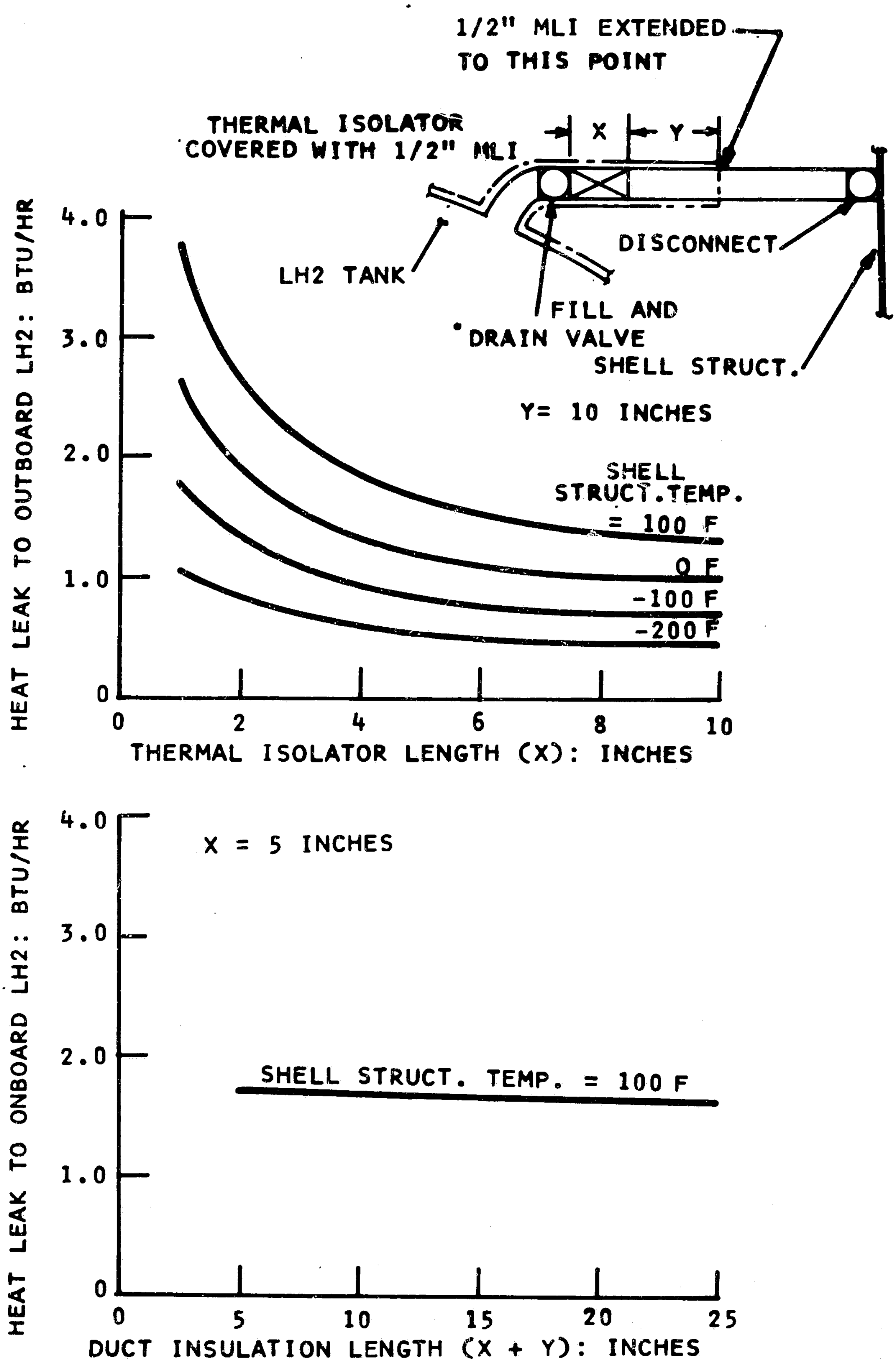


Figure 4.2-11 Heat Leak to the LH_2 Tank Thru Each Fill and Drain Line Thermal Isolator During Flight in Space

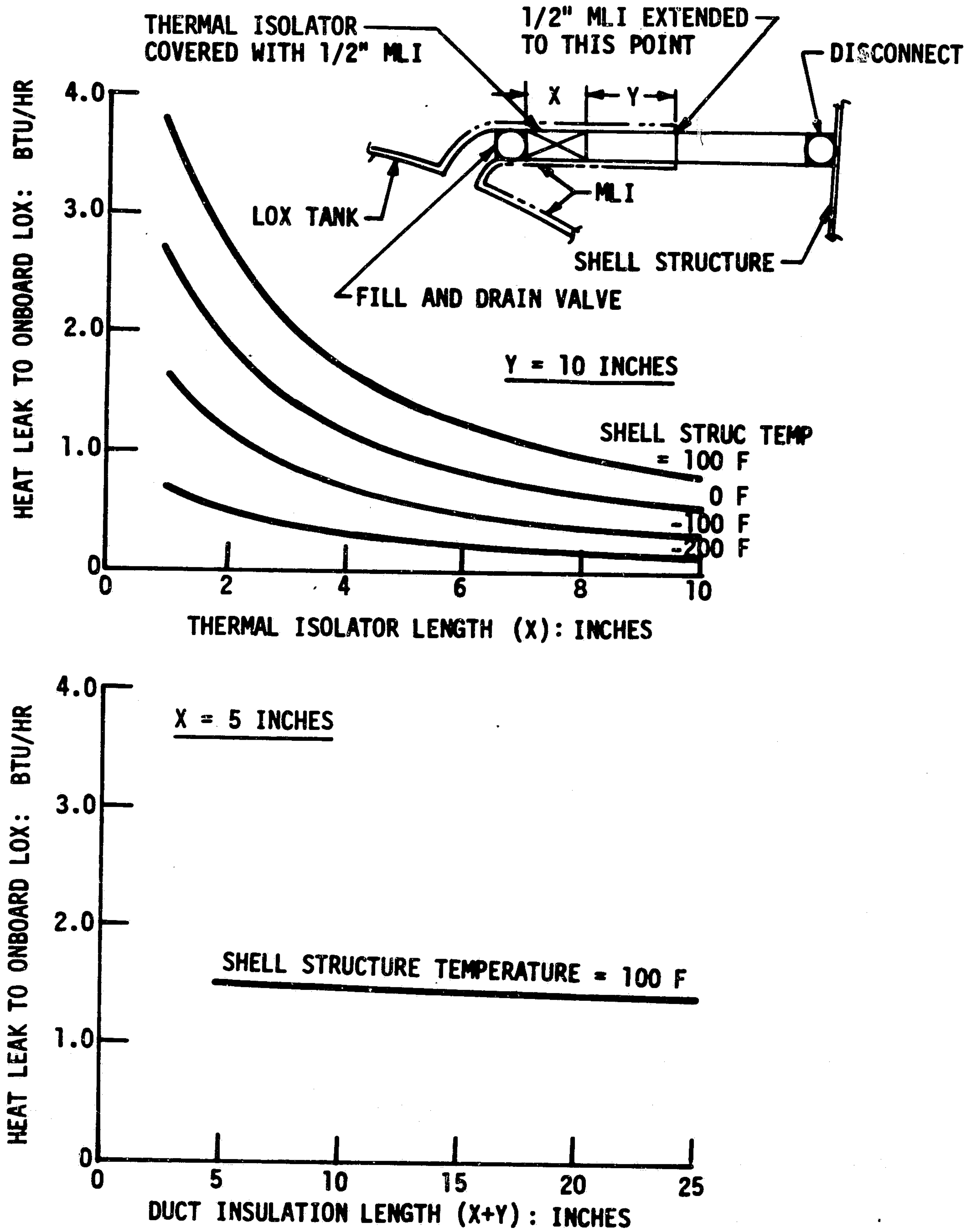


Figure 4.2-12 Heat Leak to the LOX Tank thru Each Fill and Drain Line Thermal Isolator During Flight in Space

rates for the LH_2 and LOX vent lines are about 0.035 and 0.055 pounds per hour, respectively. The total calculated boiloff for the mission between TUG ignition and redocking is approximately 12 pounds. This total boiloff does not appear to be critical to the TUG mission. If the ultimate insulation design of the vent lines would result in excessive heat leak and boiloff, then valves would have to be installed to prevent the lines from filling with propellant during orbit.

Conclusions

Heat leakes through penetrations can be reduced to an acceptable level by proper design and use of low conductivity materials. The tank supports and thrust structure when constructed of a tubular design and made of materials such as fiberglass epoxy or boron epoxy will contribute only a small percentage to the overall heat leak. The vent and fill and drain lines will require external MLI and heat blocks in the lines to minimize heat leak.

4.2.3 Propellant Feed System Heat Leak Analysis

Estimates of the heat leak rates to the propellant feed lines are required to assist in designing the thermodynamics vent system. The purpose of the thermodynamic vent system is to remove an incoming heat to the feedlines in order that a sufficient net positive suction head (NPSH) of the propellants can be maintained while the main engine is operating in the idle mode during flight. The entire feed lines are assumed to be insulated with 1/2 inch of MLI.

The heat leak calculations were made by dividing the feed lines into two areas; i.e., the area forward of the thrust structure and the area aft of the thrust structure (see Figure 4.2-13). This division of the feed lines was necessary because the feed lines are subjected to two different environmental conditions. The section of the feed lines in the area forward of the thrust structure will not be exposed to a direct solar heating. However, the section aft of the thrust structure will be subjected not only to direct solar heating but also to heat transferring from the main engine.

The calculated heat leak rates to the propellant feed systems are summarized in Table 4.2-4. The heat leak rates are for nominal heating conditions. The following detailed discussion should be referred to the local high heating conditions.

Feed Lines Forward of the Thrust Structure

The heat leak rates to the propellants in the feed lines forward of the thrust structure are presented in Figure 4.2-13 as functions of the shell structure and thrust cone temperatures. The shell structure and thrust cone temperatures in space are strongly influenced by the solar heat. The maximum solar heating environment occurs when the Tug is flying with its broadside to the sun and not rotating. The temperature of the shell structure with a high solar absorbtivity coating would be in the 200 F range, but with a low solar

NOTE:

1. THE PROPELLANT TANKS ARE INSULATED WITH A 1/2 INCH OF NARSAM MULTI-LAYER INSULATION
2. THE FEED LINES ARE INSULATED WITH A 1/2 INCH OF NARSAM MULTI-LAYER INSULATION
3. THE LENGTHS OF THE LH_2 AND LOX FEED LINES FUEL OF THE THRUST STRUCTURE WERE ASSUMED 16.8 FEET AND 2.7 FEET, RESPECTIVELY

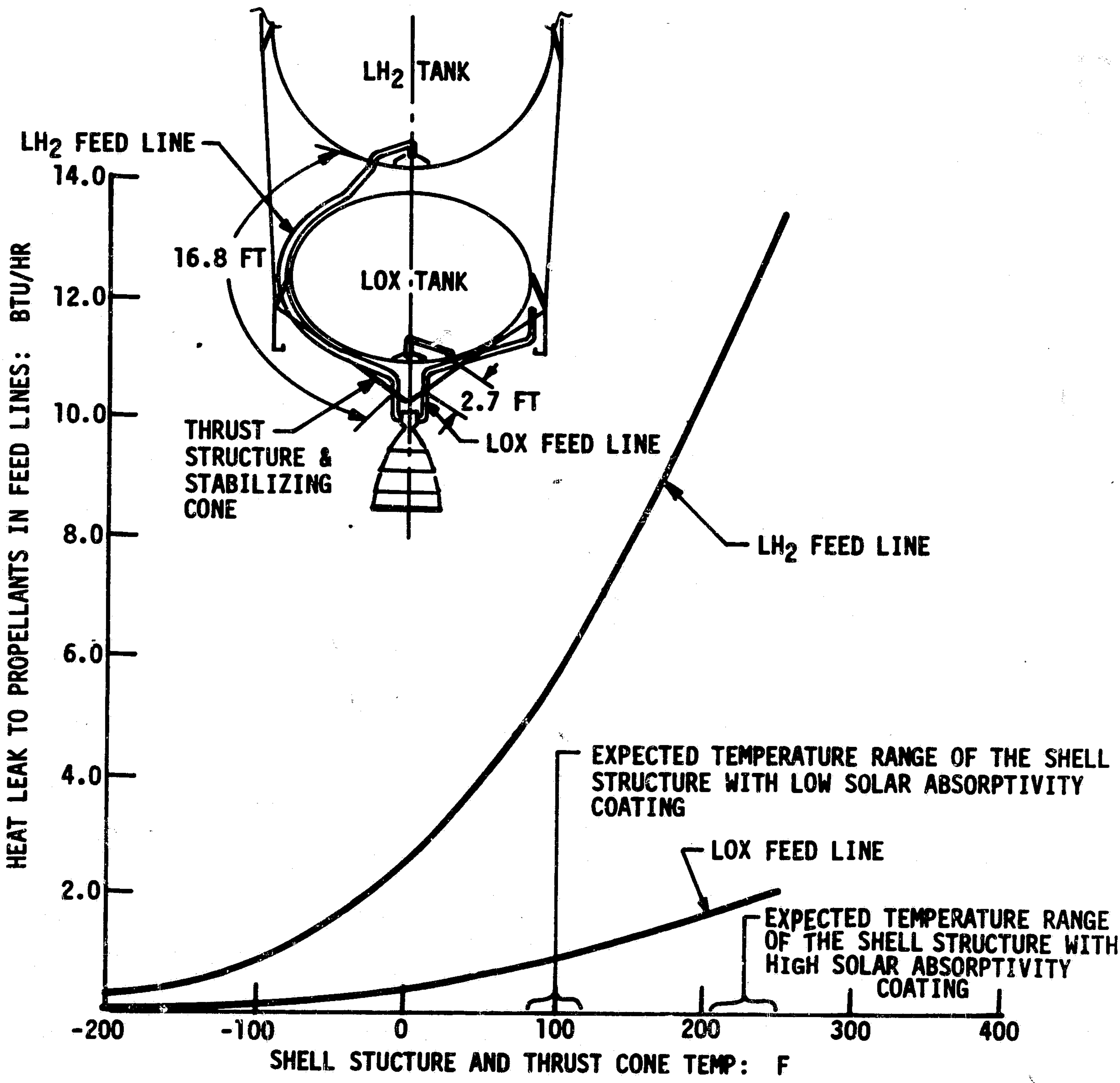


Figure 4.2-13 Heat Leaks to the Propellants in the Feed Lines FWD of the Thrust Structure During Flight in Space

Table 4.2-4. Heat Leak Rates of the Propellant Feed Systems

	LH_2 Feed System	LOX Feed System
Feed Line Fwd of [1] Thrust Cone	BTU/Hr 6.0	BTU/Hr 1.0
Prevalve [3]	1.0	1.0
Thermal Isolator [2]	4.5	3.0
Subtotal	11.5	5.0
Contingency (10%) to account for heat leaks thru brackets, supports, instrumentation, etc.	1.2	0.5
Total	12.7	5.5

Note:

[1] - Based on the local shell structure temperature of 100°F.

[2] - 6 inches long thermal isolator section is considered for both propellant feed systems

[3] - Estimates based on 100 sq. inches of wetted area. The prevalves are assumed to be insulated with a 1/2 inch of MLI.

absorbtivity coating it would be in the 80 to 120 F range. These temperatures are for the shell structure in direct sun. On the shady side, the shell structure temperature would be below -350 F. The thrust cone temperature variation would be about the same as those of the shell structure.

Based on the above mentioned shell structure and thrust cone temperatures, the heat leak rate for the LH_2 feed line is between 6 and 13 Btu/Hr and for the LOX feed line is between 1 and 2 Btu/Hr referring to Figure 4.2-13. These heat leak rates represent the maximum values since they are for the maximum solar heating condition with the space Tug flying in a non-rotating mode. If the space Tug were rotated slowly about its center axis, the shell structure temperature would be about -50 F. In this case, the heat leak rates for the LH_2 and LOX feed lines would be about 1.8 Btu/Hr and 0.3 Btu/Hr, respectively.

Feed Lines Aft of the Thrust Structure

The subject feed line systems are illustrated in Figure 4.2-1. As is shown, the thermal isolators are incorporated in the lines immediately downstream of the propellant prevalves to minimize heat transferring from the

main engine. The considered thermal isolator configuration consists of 3 mil CRES foil reinforced with 4 layers of 0.01 inch thick fiberglass epoxy. The entire feed line systems are insulated with 1/2 inch of MLI.

The calculated heat leak rates through the thermal isolators of the LH_2 and LOX feed lines are presented in Figures 4.2-14 and 4.2-15, respectively. The rates are given as functions of the thermal isolator lengths and the temperatures of the thermal isolator at the end where it is connected by a short duct to the main engine. The following assumptions were made in the heat leak calculation. First, it was assumed that the feed lines downstream of the prevalves are completely empty of propellants and filled with gas evaporated from the respective propellant. Also, the pressure of the residual gas was estimated to be in the 0.1 mm Hg range, and not lower because of the long, small cross section path the gas must travel to reach the open space. In this pressure range, a thermal conductivity of the residual gas is in a continuum flow regime, where a gas thermal conductivity is the same as that of 1 atmosphere and is independent of pressure.

The temperatures of the thermal isolators at the engine end are difficult to estimate during this study phase due to lack of necessary information. The isolator end temperatures will reach maximum values sometime after the main engine shutdown due to heat soakback from the engine components. Then the isolator end temperatures will gradually reduce until steady state has been reached. It is estimated that the steady state temperatures will be lower than 0°F for the LH_2 feed line thermal isolator end and 50 F for the LOX feed line thermal isolator end. These steady state temperatures were estimated for 6 inch long thermal isolators and it was assumed that at least 6 inches of the ducts between the thermal isolators and the main engine are insulated with 1/2 inch of MLI. The 6 inch long thermal isolators have been selected based on the propellant feed system design consideration, installation problems, and comparison of the thermal isolator effectiveness for various lengths which are shown in Figures 4.2-14 and 4.2-15. Referring to these two figures, the maximum heat leak rates through the 6 inch long LH_2 and LOX feed line thermal isolators are about 4.5 Btu/Hr and 3 Btu/Hr, respectively. These heat leak rates are based on the estimated maximum steady state temperatures of the thermal isolators at the engine end.

4.2.4 Avionics Thermal Control System

The desired thermal control system will be passive, according to study ground rules. "Passive" includes thermal control coatings, insulation, insulating spacers, and sun shields. Heaters, louvers, and heat pipes are also considered passive, although heat pipes will not be used because of their weight penalty (as per NASA directive).

The approach necessary to design a passive control system includes: (1) determination of the "worst-case" hot and "worst-case" cold external environments, (2) selection of a thermal control system that will work under worst-case hot conditions with the equipment operating at full power, and (3) determination of the required insulation, heaters, etc., that will be needed for worst-case cold conditions with the equipment operating at minimum

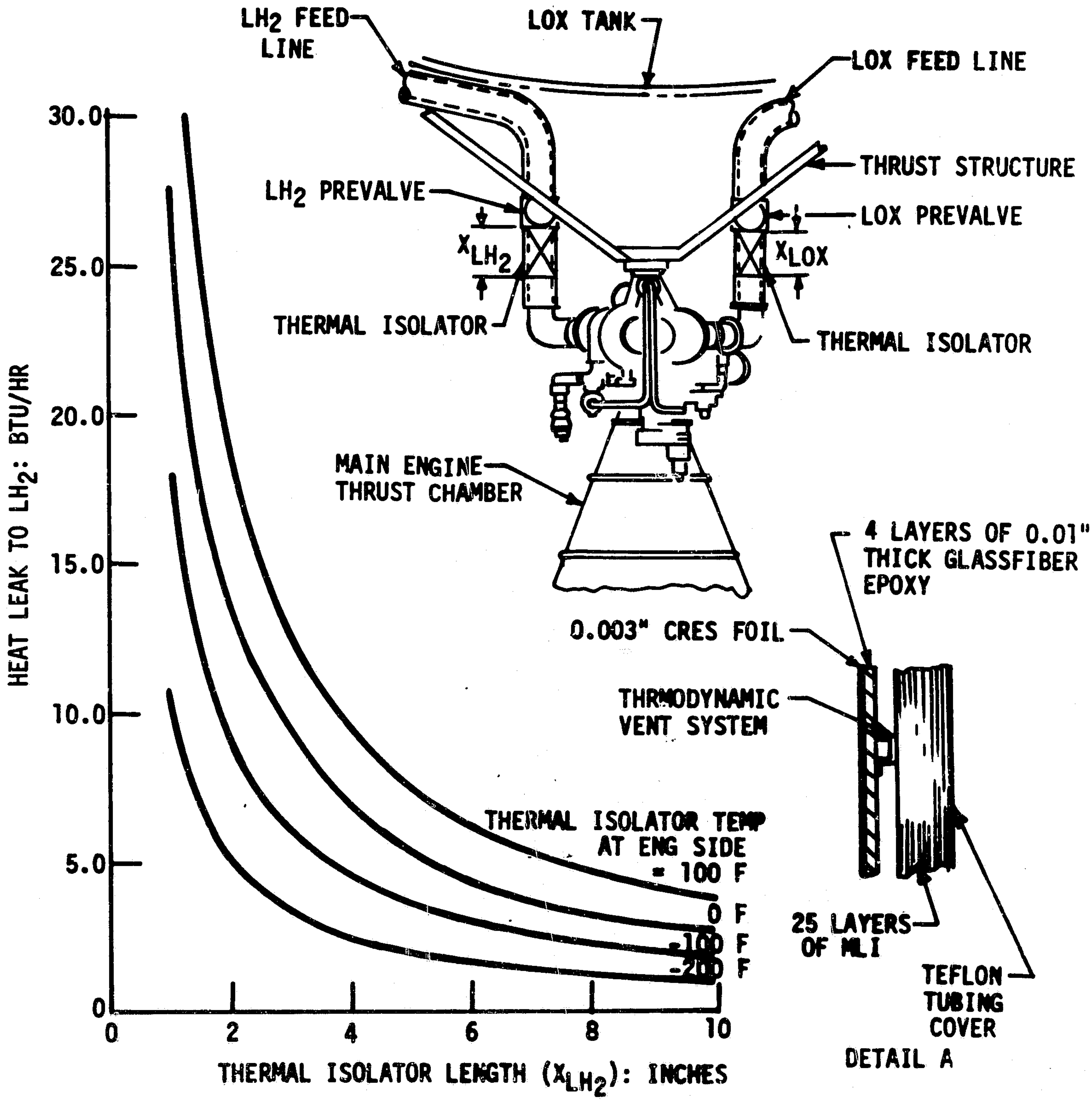


Figure 4.2-14 Heat Leak thru the LH_2 Feed Line Thermal Isolator During Flight in Space

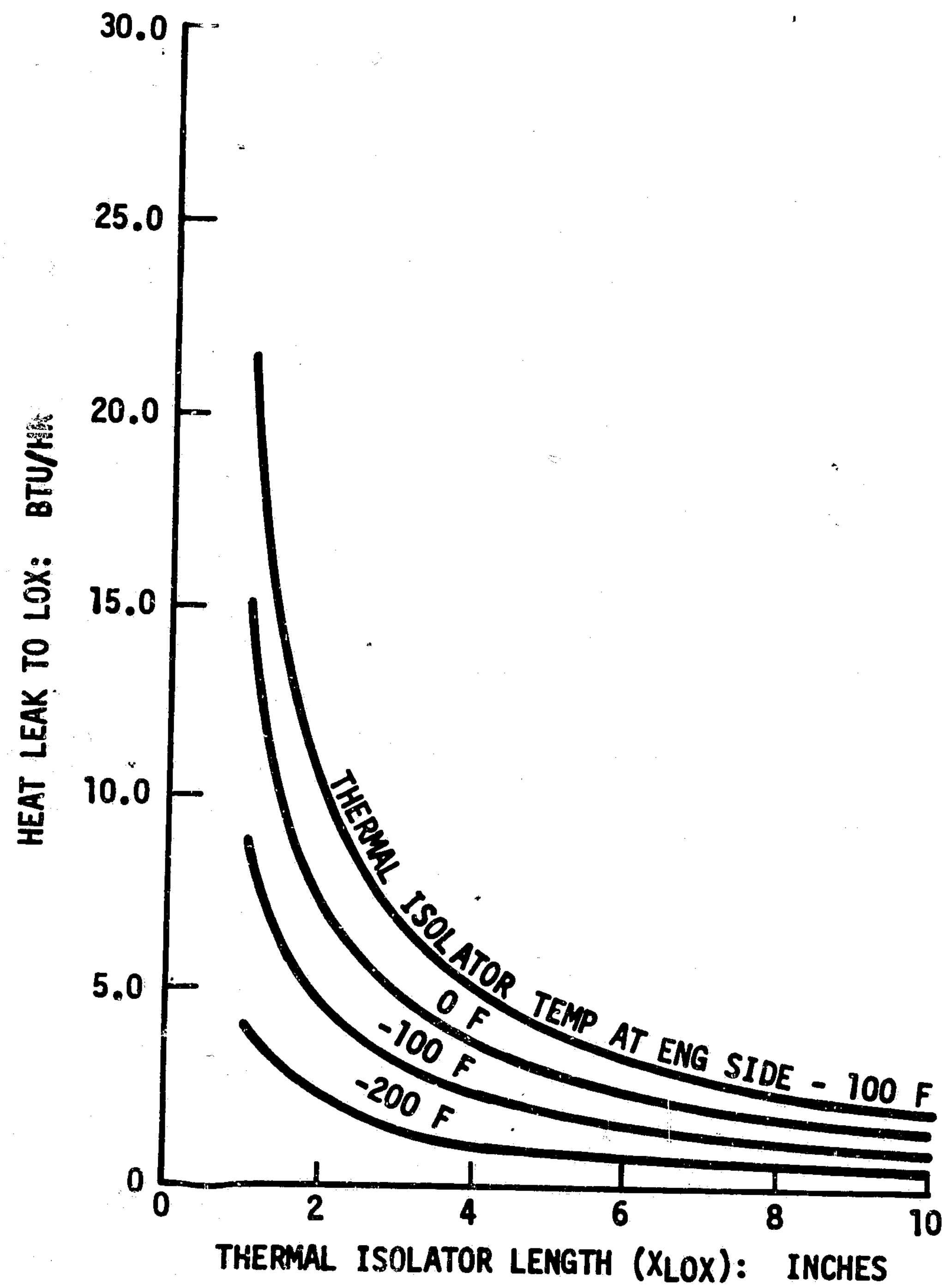


Figure 4.2-15 Heat Leak Thru the LOX Feed Line Thermal Isolator During Flight in Space

power. For the hot case, the equipment will run near the top of its allowable operating temperature range; conversely, for the cold case, the equipment will run near the bottom of the allowable temperature range. (One disadvantage of a passive system is the wide temperature range over which the equipment must operate.)

The avionics thermal control system is shown schematically in Figure 4.2-16.

Environmental Heat Loads

A number of cases were studied to determine the warmest environment that will be encountered during Tug flight. A simplified Radiosity network of the forward skirt area was solved for various combinations of solar incidence angles and thermal control coating properties. These are presented in Table 4.2-5. Cases 1-4 assume the presence of a solar reflecting shield over the open forward end of the vehicle. Cases 5-8 assume no such shield. It can be seen that a solar reflecting shield is a very desirable item, in the absence of a payload, in the vicinity of the avionics equipment. The cavity formed by the hydrogen tank and forward skirt constitutes a solar "trap" which can conceivably reach high temperatures upon direct solar radiation even if all internal surfaces are white (case 7). The probability is that these surfaces will not or cannot be all white. In this case, local temperatures can approach 200°F with subsequent failure of a passively controlled thermal control system (case 8).

If it is assumed that a sun shield will be used for protection when the Tug is not docked to the payload, then the "worst case hot" situation will occur when the payload is attached and the sun is vertical to the X-axis (Figure 4.2-17). Assuming an α/ϵ equal to 0.3, the average structural environmental temperature was estimated to be 60°F. This is adequate for most of the avionics equipment, which can operate at 120°F if necessary. However, the IMU environment is limited to 80°F, and a coating with properties closer to 0.20 - 0.25 is required in areas adjacent to this component. Values as low as this have been reported in Lunar Orbiter studies and elsewhere and certainly are not beyond present state-of-the art capability. Thermal coating degradation should not be a problem on Tug because of the short 7-day mission and the fact that low-energy protron environment will not be encountered. Ultraviolet resistant coatings such as Z-93 and S-13G should not degrade significantly. Therefore, it was assumed that the α/ϵ ratio will be 0.25 in the vicinity of the IMU, resulting in an estimated structural temperature of 35°F, which is adequate for IMU cooling. If problems are still encountered, the local usage of optical solar reflectors (OSR's) can be utilized.

The worst-case cold environment will occur when the Tug is oriented tail-to-the sun and all forward areas will view deep space (Figure 4.2-17). Structure temperatures for this case were assumed to be -300°F. Heater requirements were based on this environment, with the equipment operating at minimum power levels.

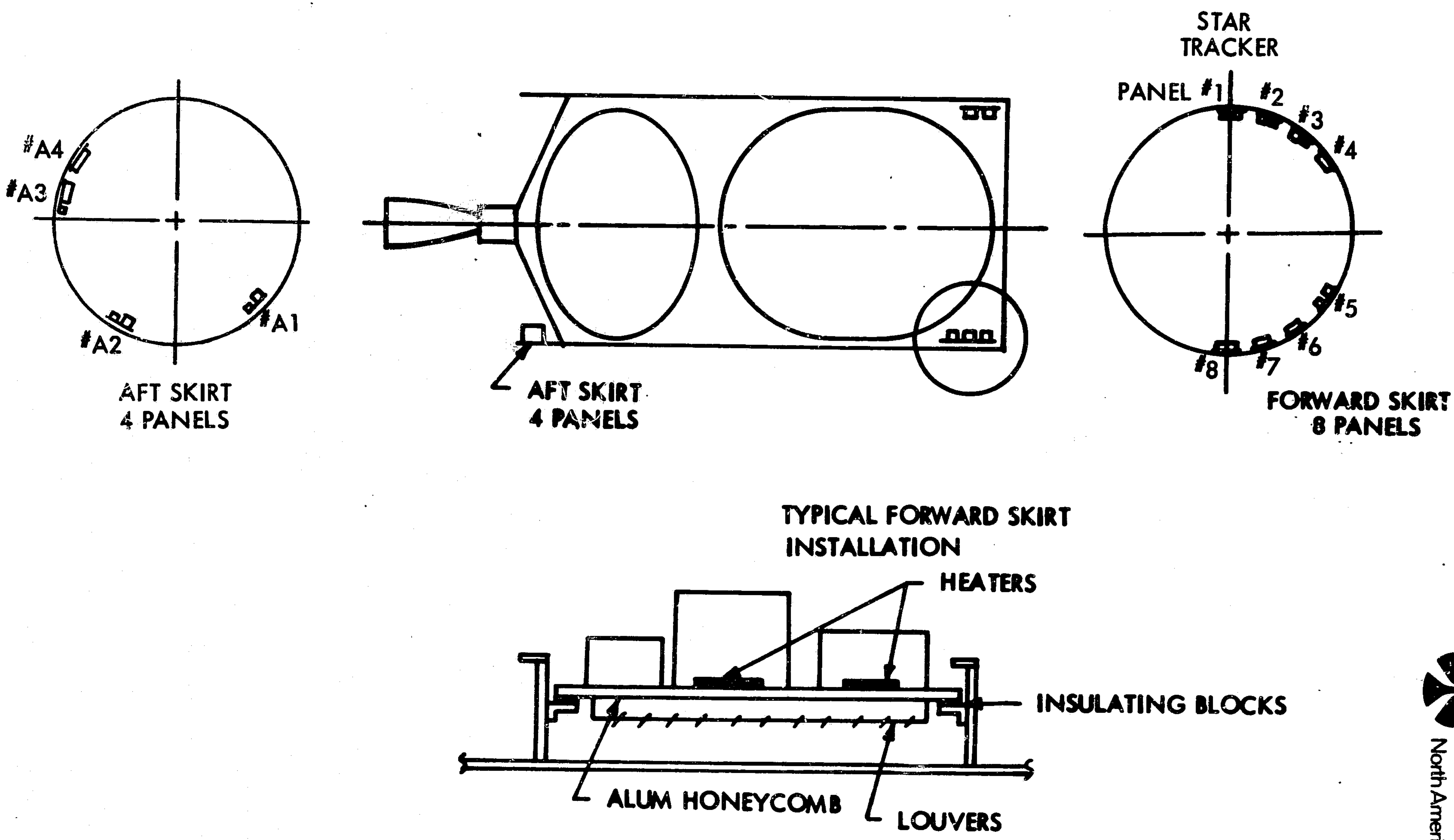
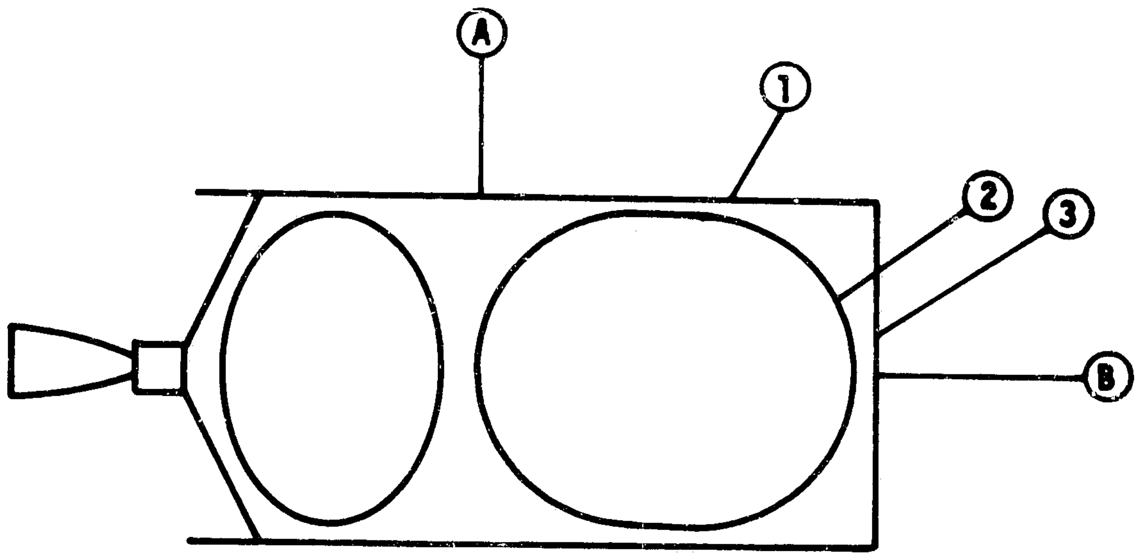


Figure 4.2-16 Avionics Thermal Control Configuration

Table 4.2-5 Structural Environmental Temperatures, Forward Skirt



CASE	α / ϵ	α / ϵ INTERNAL	α_3 / ϵ_3	TEMP F t_1	t_2	t_3
SUN AT A, NO PAYLOAD, WITH SUN-SHIELD						
1	0.3/0.9	0.8/0.8	0.3/0.9	55.4	15.9	-57.6
2	0.9/0.9	0.8/0.8	0.9/0.9	218.4	166.5	69.7
SUN AT B, NO PAYLOAD, WITH SUN-SHIELD						
3	0.3/0.9	0.8/0.8	0.3/0.9	-105.4	-27.3	35.2
4	0.9/0.9	0.8/0.8	0.9/0.9	6.5	109.3	191.5
SUN AT A, NO PAYLOAD, NO SUN-SHIELD						
5	0.3/0.9	0.8/0.8	-	37.4	-41.9	N/A
6	0.9/0.9	0.8/0.8	-	194.7	90.4	N/A
SUN AT B, NO PAYLOAD, NO SUN-SHIELD						
7	0.3/0.9	0.3/0.9	-	1.8	91.3	N/A
8	0.3/0.9	0.8/0.8	-	37.8	211.4	N/A
SUN AT A, WITH PAYLOAD						
9	0.3/0.9	0.8/0.8	-	60.0	60.0	60.0
10	0.2/0.8	0.8/0.8	-	35.0	35.0	35.0

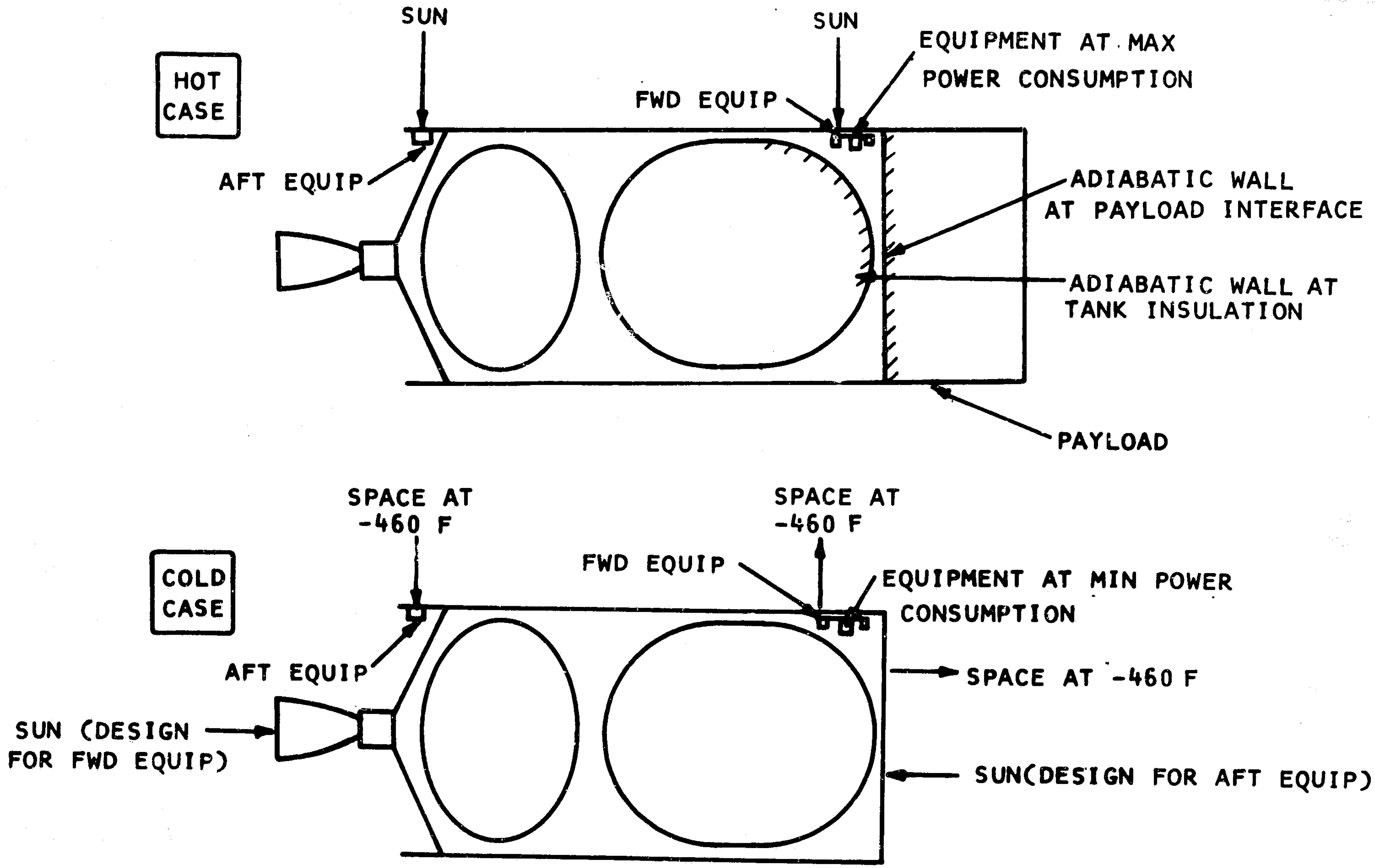


Figure 4.2-17 Worst Case Environmental Conditions

Heater Requirements

Avionics instruments in the forward skirt area will be mounted on aluminum honeycomb shelves or panels which in turn will be fastened to the frames in the Tug skin. The panels will be thermally isolated from the frames by means of non-metallic spacers. This will prevent heat loss by conduction to the structure, which ranges from +80 to -300°F during the mission. The panel and component areas will be sized such that all heat loads generated by the equipment during operation can be rejected to the vehicle skin and internal structure for the hottest anticipated environment. Results of the study indicate that very little insulation will be used in the avionics area due to the large amount of heat that must be rejected when the equipment is operating at full load under "worst-case hot" conditons.

Passive thermal control requirements for a typical case (Panel 2, forward skirt area) are shown in Figure 4.2-18. Permissable operating temperature range for the equipment mounted on this panel is 30 to 120°F. Equipment power consumption ranges from a continuous minimum of 28.5 watts to a maximum of 61.5 watts. It was assumed in the study that all power will be converted to heat. The thermal control system design point was based upon maximum allowable temperature (120°F), maximum heat generation (61.5 watts), and "worst case hot" environmental conditions. Maximum heater requirements were based upon minimum allowable operating temperature limits (30°F), minimum heat generation (28.5 watts), and worst case cold environmental conditions. Maximum heater requirements in this case were calculated to be 59.0 watts. Average heater power requirements cannot be determined until vehicle attitude and power consumption timelines are defined. It was assumed, therefore, that average heater requirements were 2/3 maximum, which for Panel 2 is 39.3 watts.

Heater requirements for all avionics equipment are summarized in Table 4.2-6. Total maximum heater requirements are 234 watts, with a total average requirement of 156 watts. Also included in this table are heater requirements necessary to maintain minimum non-operating temperatures when the Tug is docked inside the EOS. An average of 123.9 watts of heating will be required to maintain avionics temperature at -65°F, with the assumption of a -100°F EOS cargo-bay environment.

Louvers have been included as part of the forward skirt panel 4 thermal control system. These are the same type that have been used successfully on a number of space vehicles for several years. Each louver panel is made from polished aluminum blades actuated in sets of two by bimetallic thermostats. Fully closed to fully open position covers a range of 30°F, with a corresponding change in effective emissivity from 0.13 to 0.80. Operating temperature is adjustable and was assumed to be 40°F in this study, i.e., the operating range is 40 to 70°F. Louver weight is 1.5 Lb/Ft^2. Performance of the system for panel 4 is presented in Figure 4.2-19. It can be seen that 4.0 square feet of louver area will lower the required heater power by approximately 76 watts for worst case conditions. Additionally, the louvers will act as a control device and tend to keep equipment temperature within the 40 to 70°F band for nominal environmental conditions. Of course, a final decision between a heater/louver system or a pure heater system will depend

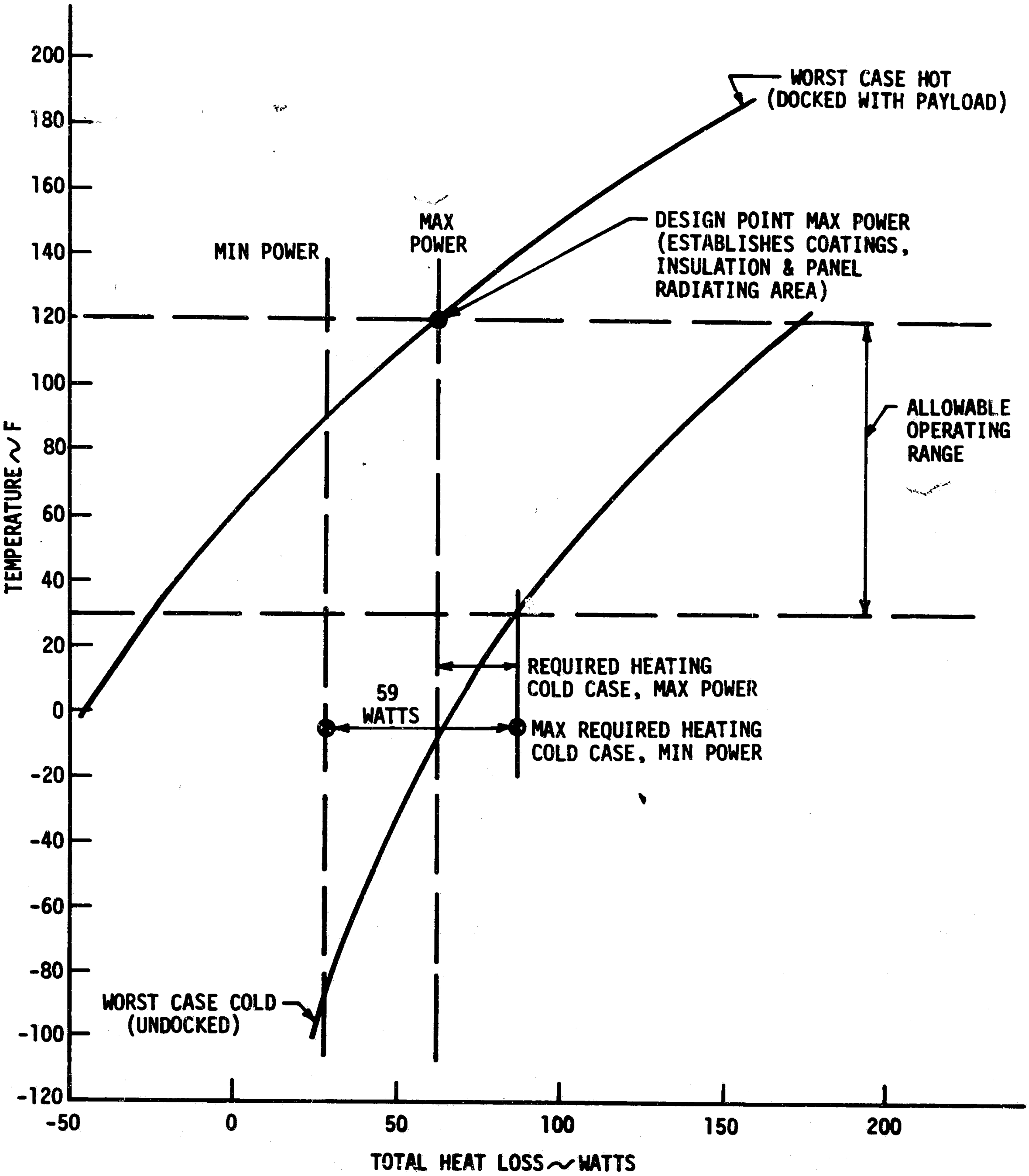


Figure 4.2-18 Typical Passive Thermal Control Requirements (Panel 2)

Table 4.2-6 Heater reRequirements

Forward Skirt	Design Temp. Range (Operating)	TUG FLIGHT: Max. Power (w/o Heaters)	TUG FLIGHT: Min. Cont. Power (w/o Heaters)	TUG FLIGHT: Max. Heat Loss	TUG FLIGHT: Max. Heater Req.	TUG FLIGHT: Ave. Heater Req.	EOS CARGO BAY: Min. Cont. Power	EOS CARGO BAY: Ave. Heater Req.
Panel #1 (ST, IMU, TV)	0-80 °F	84.6 w	72.6 w	149.9 w	0.0 w ①	0.0 w ①	0.0	15.7 w
Panel #2 (COMM, GN&C)	30-120	61.5	28.5	87.5	59.0	39.3		11.6
Panel #3 Data MGMT)	0-120	30.0	30.0	33.1	3.1	2.1		5.7
Panel #4 (COMM)	30-120	127.0	8.0	104.8	96.8	64.6		15.8
Panel #5 (Data MGMT)	0-120	90.0	90.0	99.3	9.3	6.2		17.1
Panle #6 (Data MGMT)	0-120	43.0	43.0	47.4	4.4	2.9		8.1
Panel #7 (LR&HS Elect.)	0-120	51.0	26.0	56.2	30.2	20.1		9.7
Panel #8 (LR&HS)	65 ± 20	27.0	12.0	28.2	16.2	10.8		5.8
<u>Aft Skirt</u>								
Panel #A1	0-120	70.5	48.5	58.4	9.9	6.6		10.0
Panel #A2	0-120	112.5	112.5	93.3	0.0	0.0		16.0
Panel #A3	0-120	28.5	18.5	23.6	5.1	3.4	0.0	4.1
Panel #A4	0-120	30.0	30.0	25.1	0.0	0.0	0.0	4.3
TOTAL		755.6 ②	519.6	806.8	234.0	156.0	0.0	123.9

① Up to 85 watts supplied by internal IMU Heater

② All equipment operating

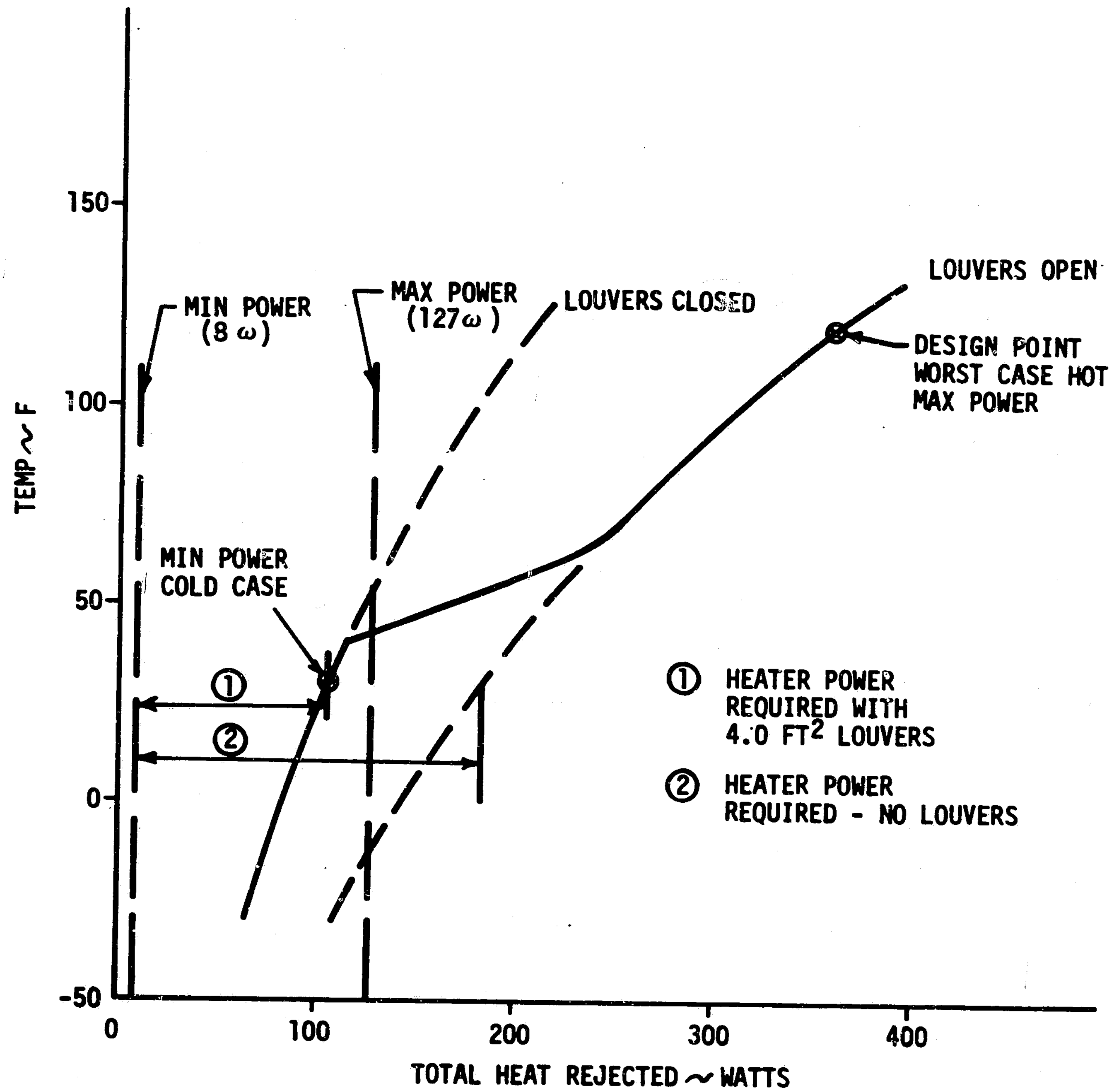


Figure 4.2-19 Panel 4 Heater Power Requirements

in large part upon available electrical power and the weight trade-off between fuel cell propellant and louver weight. Such a comparison was beyond the scope of the present study. A basic reason for the application of louvers to panel 4 was the large variation between maximum and minimum operating power. If this margin were to be reduced, then a louver system would be less desirable. It should also be noted that the temperature regulating feature of a louver system may be useful in maintaining control over temperature critical items on other panels, such as the IMU (0-80°F) and the laser radar (65±20°F).

Conclusions

In conclusion it was demonstrated that a passive thermal control system for the avionics equipment was feasible. The system requires the proper selection of thermal control coatings, insulation, panel radiating areas, louvers and electrical heaters in order to maintain required equipment temperature limits when exposed to predicted mission environments. Heater requirements were determined to be 234 watts maximum with a continuous average of 156 watts.

4.2.5 RCS Thermal Protection Requirements

The location of the RCS engine clusters on the aft skirt will result in direct plume impingement on the graphite epoxy composite structural body of the Tug. The exhaust plumes from both the forward firing pitch thrusters and side firing roll thrusters will impinge on the Tug body. Also the plume from the aft facing thrusters will impinge on the main engine. There are a total of four clusters located 90° apart. Two of the clusters have five thrusters and two clusters have aft and forward facing thrusters only.

A thermal model of the graphite epoxy composite structural body was developed for use with a thermal analyzer program for the IBM 360 computer in order to determine temperatures of the uninsulated Tug body so that the areas which require thermal protection can be defined. Plume impingement heating rates presented in the above section were used with the thermal model. It was determined from this initial analysis that the maximum allowable temperature of 350°F for the graphite epoxy composite would be exceeded in areas which are exposed to heating rates greater than 0.20 Btu/Ft^2-sec. These areas of concern are shown in Figure 4.2-20 and total approximately 46 square feet. The maximum firing durations used in the study were two minutes for the pitch jets and one minute for the roll jets.

In order to determine thermal protection requirements in the areas of concern, a thermal model was developed for the Tug body which included an external insulation system utilizing Dynaflex. Dynaflex was selected for this application since it has a relatively low density (approximately six pounds per cubic foot) with good insulating properties to approximately 2800°F. An analysis of the area of maximum heating was conducted and it was determined that a Dynaflex thickness of 0.25 inches is required to maintain a maximum graphite epoxy wall temperature of 350°F. This thickness was assumed for all areas which require protection since the total weight of Dynaflex for this

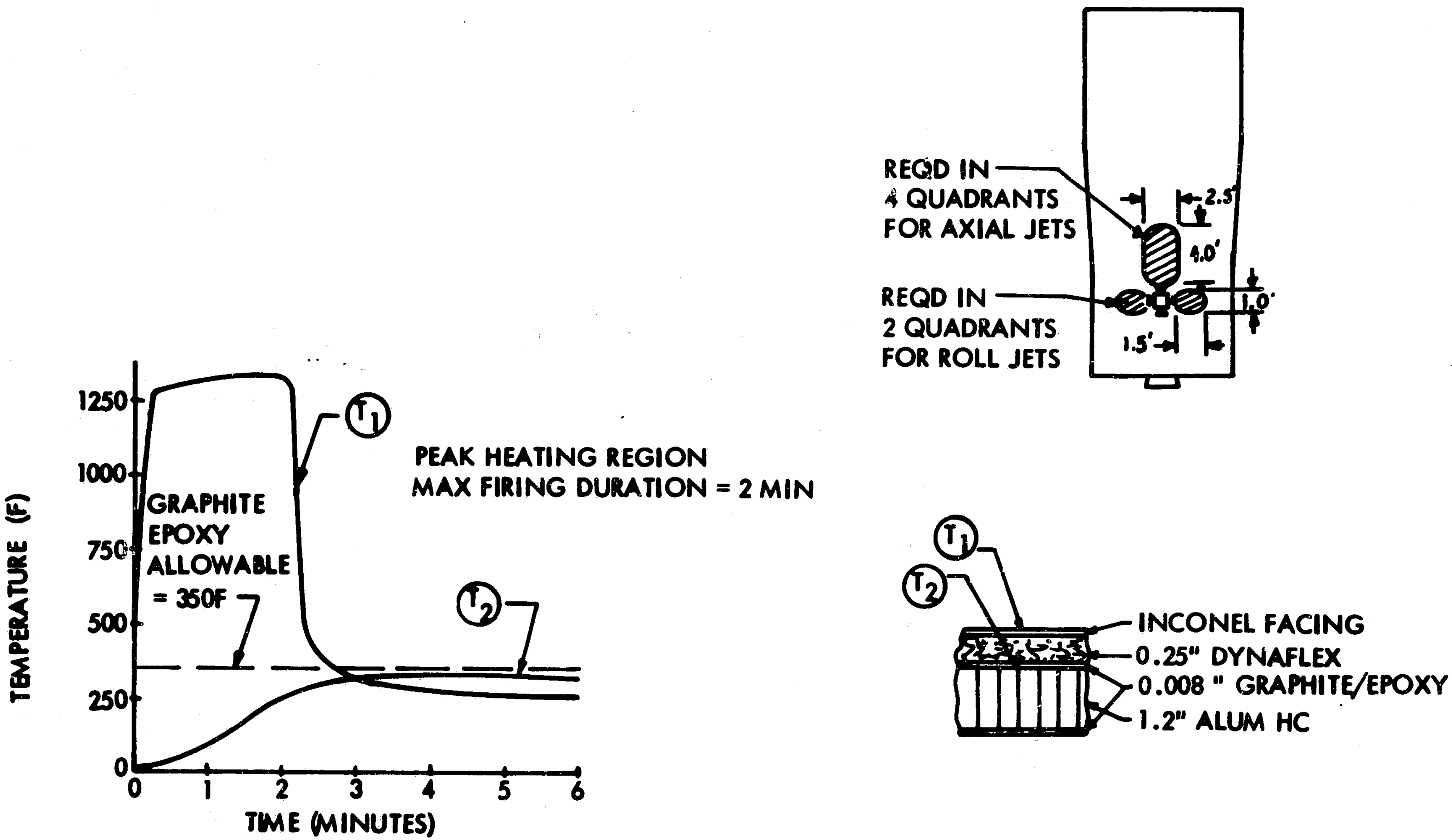


Figure 4.2-20 RCS Thermal Protection Requirements

thickness is only about six pounds. The temperature histories calculated for this configuration are shown in Figure 4.2-20.

The calculated main engine nozzle heat flux from a single aft firing APS thruster is shown by Figure 4.2-20. As indicated, the worst case maximum heat flux at the nozzle is one Btu/sec-ft^2. Resulting predicted nozzle temperatures are shown in Figure 4.2-21 with a maximum of 690°F calculated at the end of a two minute firing. The temperature effects analysis was addressed to the following areas of concern with the tentative conclusions indicated.

The possibility exists that the nozzle heating may cause reduced strength or introduce distortion and/or thermal stresses. The use of inconel or a similar steel material with dump cooling through axial channels welded to a removable nozzle extension has been proposed for the baseline engine. Such a design and material is predicted to tolerate temperatures to 750°F and any resulting circumferential differential heating. The nozzle will be structurally stiffened to accommodate relatively severe load reversals and vibration conditions which occur during firing. Therefore, the nozzle will be designed to accommodate the stresses associated with this heating condition.

The RCS heating of the engine nozzle was reviewed in connection with possible affect on the chill down cycle. Since there is no purpose for chilling the nozzle itself and the heat transfer resistance from the nozzle to the pump assembly it has been determined that any affect on engine chill down is insignificant.

The desirability for a special nozzle coating because of the RCS engine gas impingement was considered. Nozzle surface treatment will normally be required to resist corrosion and to attain satisfactory optical properties to assure reasonable space equilibrium temperatures. Use of a grey oxidized surface attained by heating the nozzle in air is tentatively planned. Such a surface provides an emissivity of about 0.8 which is the value used to calculate the temperatures of Figure 4.2-21. It is concluded that no special coating is needed.

Conclusions

Thermal protection from RCS plume impingement heating will only be required in local areas, totaling about 46 square feet. About 0.25 inches of a lightweight, low conductivity insulation such as Dynaflex will protect the graphite epoxy Tug structural body. No thermal protection will be required on the main engine.

4.2.6 Base Region Thermal Protection

A thermal analysis was made to determine temperatures of the aft meteoroid shield during main engine firing. The maximum predicted radiation heating rate of 0.0004 Btu/Ft^2-Sec from the exhaust plume was used along with the maximum solar heat rate of 0.124 Btu/Ft^2-sec to predict a maximum shield temperature of approximately 65°F. This temperature is based on use of a ultraviolet

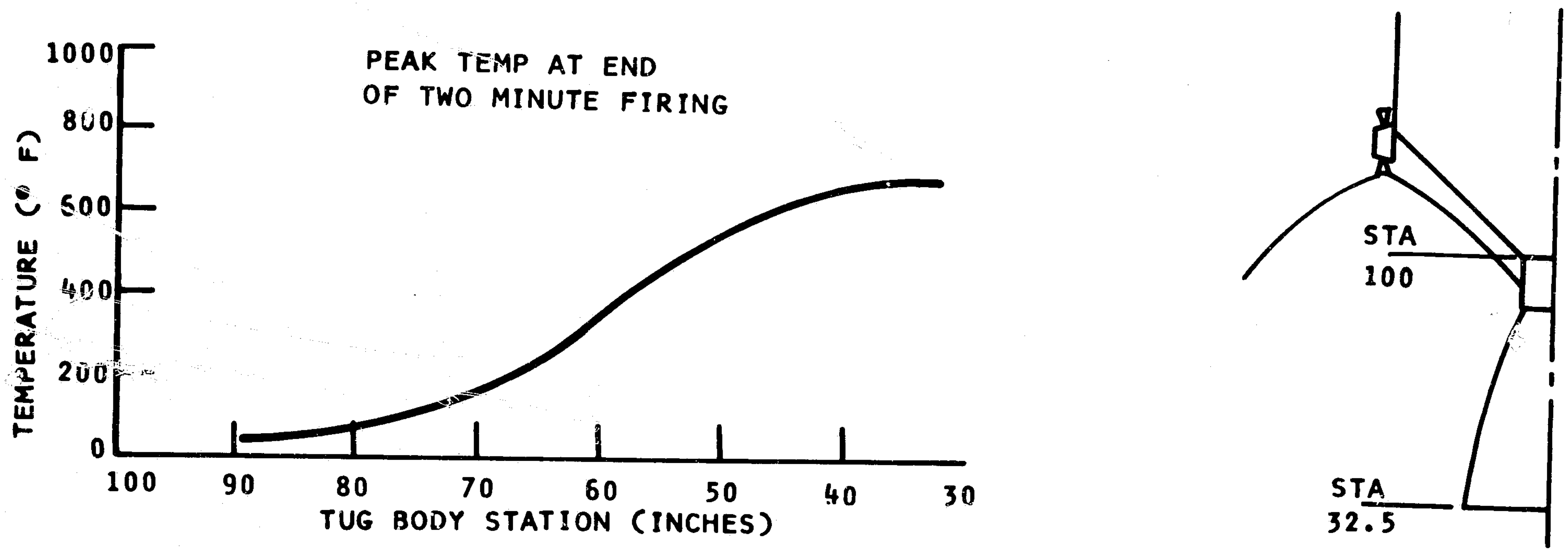


Figure 4.2-21 RCS Plume Impingement on Main Engine

resistant white paint such as S-13G with α/ε ratios of 0.25 to 0.30. The shield is made of two basic sections; one constructed of reinforced fiberglass epoxy and the other of a graphite epoxy composite both of which have allowable temperatures of 350°F. Since the predicted maximum meteoroid shield temperature for the worst case heating condition is less than the allowable temperature, the shield is also adequate for providing base region thermal protection.

4.2.7 RCS Accumulator Tank Insulation Analysis

The RCS gaseous hydrogen (GH_2) and gaseous oxygen (GOX) tanks must be maintained at a nominal -260°F and -60°F temperature level throughout the Tug mission lifetime. A study was undertaken to determine thermal requirements for these items. Results indicate that a passive system consisting basically of insulation and thermal control coatings will suffice if periodic GH_2 and GOX replenishment is maintained in sufficient quantity to counteract the heat leak into or out of the tanks. Multilayer insulation will be required with a nominal 30 layer thickness. In addition, an outer cover layer or sun shield with a solar reflecting (white) surface will be required to prevent overheating in the case of direct solar irradation.

Space Flight

Both the GH_2 and GOX tanks experience steady outgoing flow rates due to fuel cell operation and occasional RCS firing. It was assumed in the study that the GH_2 tank outflow rate was 0.111 Lb/Hr and the GOX tank outflow rate was 0.889 Lb/Hr. Propellants were assumed to be replenished every four hours at 0.5 Lb/Sec (GH_2) and 1.5 Lb/Sec (GOX). Incoming propellant temperatures were assumed to be -260°F (GH_2) and -60°F (GOX). The basic approach in selecting a passive thermal control system for these tanks was to utilize the sensible heat contained in the replacement propellant to make up the heat gained or lost by the propellant in the tank during the previous four hours of operation. If replacement propellant were not periodically available at the desired temperatures, then a passive system could not be relied upon because the tanks and their contents will eventually reach ambient temperature regardless of the efficiency of the tank insulation. Inasmuch as the incoming and outgoing flow are small, a multi-layer "superinsulation" will be required to minimize heat transfer through the tank walls. Foam insulation requirements were investigated and were found to be excessive, approximately 1.0 ft thick for the GOX tank and 2.0 ft thick for the GH_2 tank.

A summary of tank temperature requirements and insulation requirements is presented in Table 4.2-7. External environmental structural temperatures around the tanks were assumed to be +60°F for the worst case hot condition and -300°F for worst case cold. The allowable temperature excursions presented in Table 4.2-7 are +60 and -25°F for the GH_2 tank and +50 and -25°F for the GOX tank. These will allow hot or cold soak for an indefinite time if the propellants are replenished periodically. The temperature-time history for the GH_2 tank for continuous hot conditions is shown in Figure 4.2-22. It can be seen that the response is fairly slow which indicates that the above temperature limits (based on steady-state conditions) are conservative.

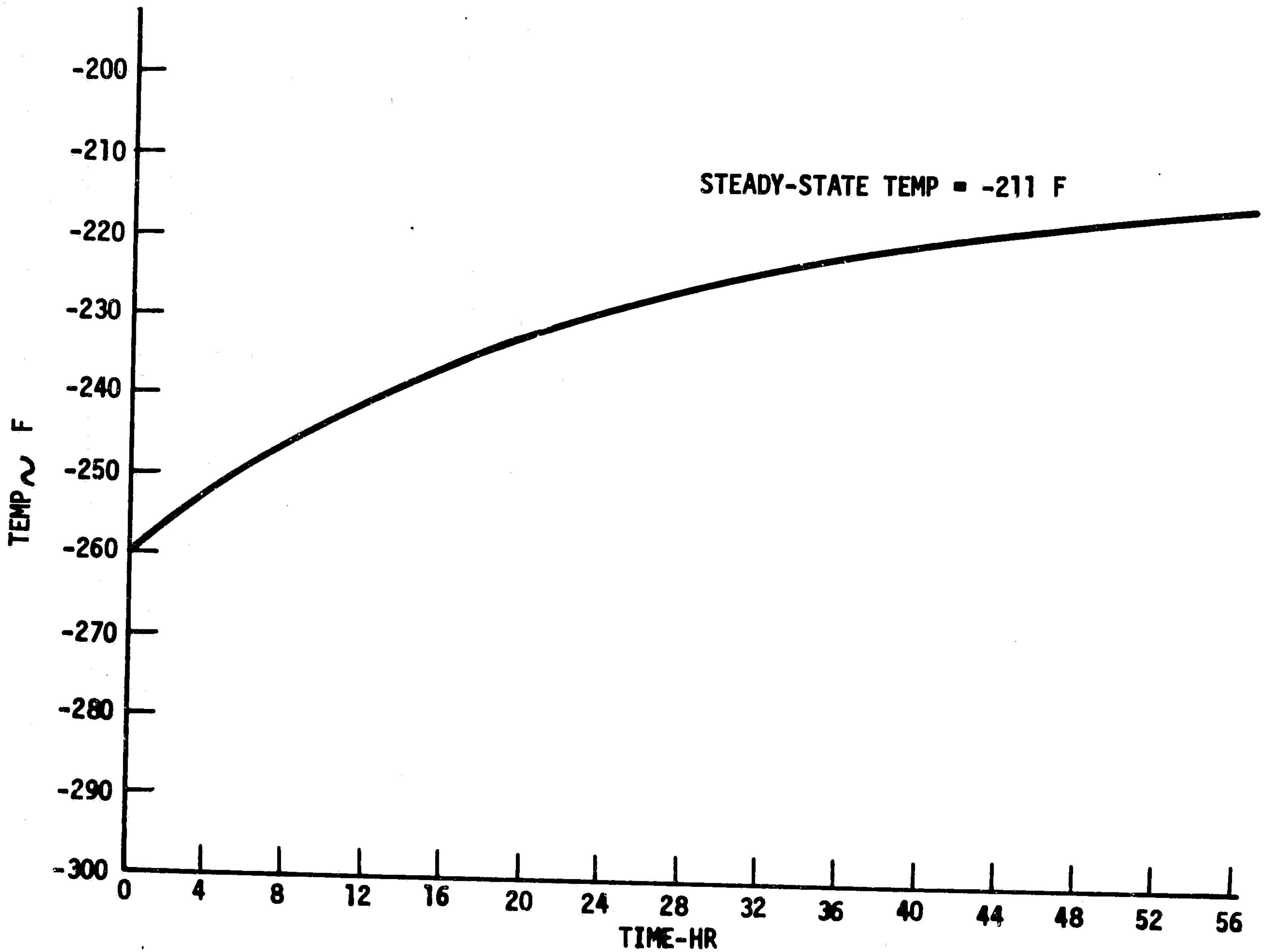


Figure 4.2-22 GH_2 Temperature Rise for Continuous Worst-Case Hot Conditions

Table 4.2-7. RCS Tank Summary

Tank	Temperature Requirements	Insulation Requirements	
		Type	Thickness
GH_2	-260^{+60}_{-25} °F	30 Layers MLI	0.75" nominal (0.5" Local)
GOX	-60^{+50}_{-25} °F	30 Layers MLI	0.75" nominal (0.5" Local)

In order to prevent serious heat leaks into or out of the tanks, both the inlet and outlet lines must be insulated. The inlet lines should be insulated with MLI for the entire distance between the heat exchanger and the tank. The outlet should be insulated with MLI for at least 1.0 foot, but preferably more, perhaps 2.0 feet.

Pre-Launch and Launch

An investigation was made to determine whether MLI with a soft cover will be sufficient for ground operation or if an evacuated dewar system will be necessary. Obviously, once in space MLI will perform the same in either case. If a vacuum insulation were required during pre-launch and launch, then a dewar system would be necessary. However, it appears that MLI with a soft cover will function on the ground if purged with nitrogen and a continuous circulation of GH_2 and GOX is used to absorb the increased heat transfer through the insulation and thereby maintain required temperatures. Up to 0.05 Lb/sec of GH_2 and 0.10 Lb/sec of GOX are available, which is more than enough.

During and after boost, MLI must depressurize to at least 10^{-4} TORR within approximately 0.5 hour to minimize heat transfer. Any condensation or ice formation may prolong this time. Therefore, prior to launch, a dry nitrogen atmosphere should be maintained, otherwise a dewar system may be necessary.

Conclusions

The results indicate that a passive system consisting basically of insulation and thermal control coatings will be adequate if periodic GH_2 and GOX replenishment is maintained in sufficient quantity to counteract the heat leak into or out of the tanks. About 0.5 inches of MLI will be required. In addition, an out cover or sun shield with a solar reflecting (white) surface will be required to prevent overheating in the case of direct solar radiation.

4.3 GUIDANCE AND CONTROL

4.3.1 Guidance, Navigation, and Control Concepts

The guidance, navigation, and control (GN&C) concepts are essentially the same as those delineated by the Project Directive and are presented in Figure 4.3-1. Detailed concepts are tabulated in Table 4.3-1 along with the GN&C events involved and the needed or affected hardware.

4.3.2 Assumptions and Ground Rules

The following assumptions and ground rules were used in determining the guidance, navigation, and control requirements.

1. State vector of Target/Shuttle is more precisely known than the Tug, and is available (via data link) prior to injection burns.

2. Target/Shuttle is installed with Laser reflector.

3. GN&C accuracy requirements are:

   Velocity - 5 m/sec (16.4 Ft/Sec)

   Altitude - Low orbit 10 Km (32,800 Ft)
   Synch. orbit 50 Km (164,000 Ft)

   Docking:

Position	± 0.75 Ft.
Velocity	± 0.1 ft/sec
Attitude	0 - 1 deg
Attitude Rate	0 - 0.5 deg/sec

4. Navigation accuracies achievable with star and horizon sensor data are equal to those reported in Reference 4-1 and Reference 4-2.

5. During TVC, the acceptable angular control acceleration is 1 deg/sec/sec.

6. The MPS thrust is nominally 10,000 lb.

7. Fail-safe criterion for attitude control in vicinity of Shuttle vehicle.

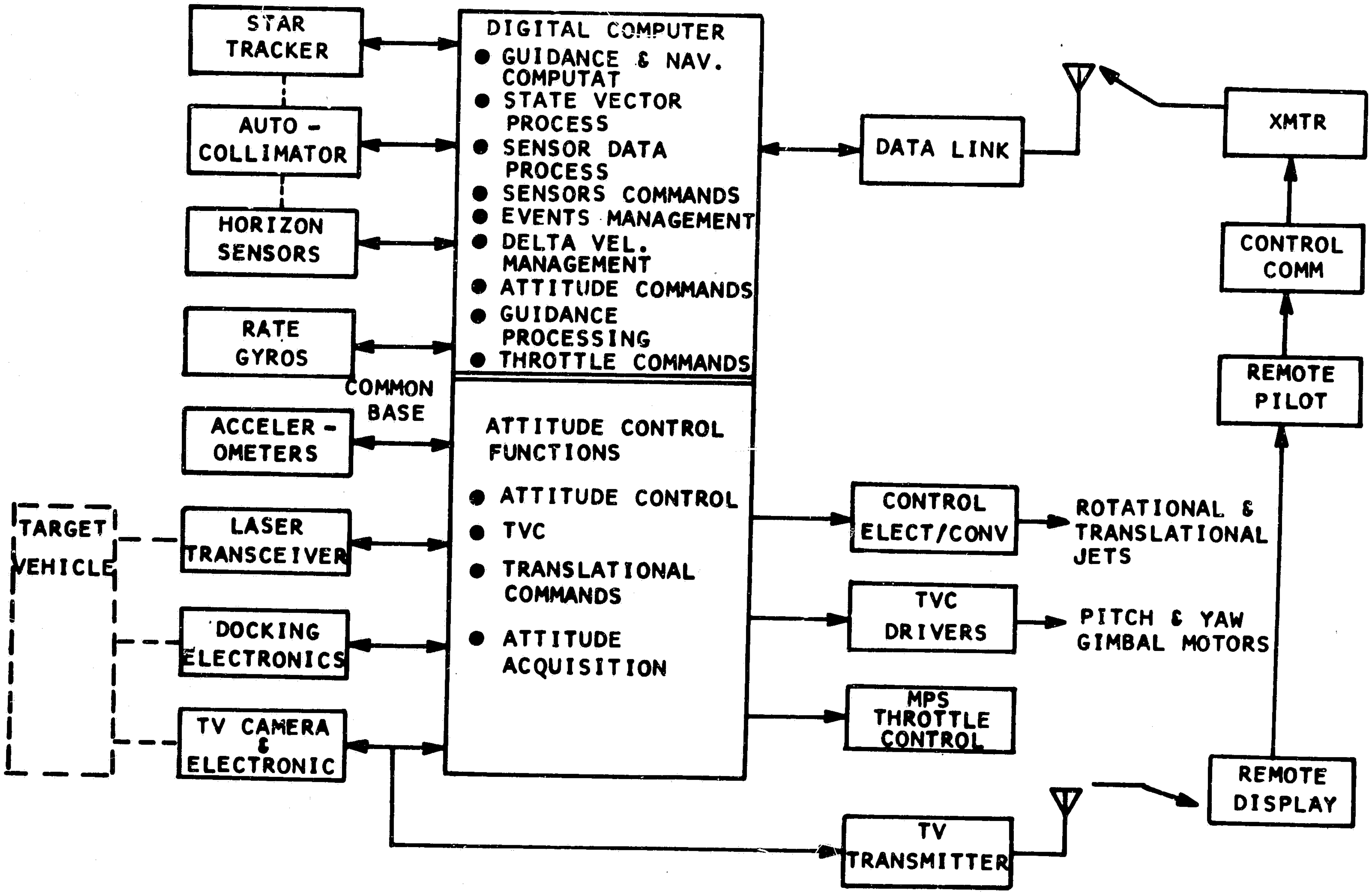


Figure 4.3-1 Guidance, Navigation, and Control Block Diagram

Table 4.3-1 Proposed Guidance, Navigation and Control Concepts (Sheet 1)

GN&C EVENTS	FUNCTIONS	CONCEPT PROPOSED	HARDWARE NEEDED OR AFFECTED
Prior to Separation From Shuttle Obtain own and target state vectors from Shuttle Obtain Attitude and Rate Data from Shuttle Start up navigation system and computations Compare own attitude and navigation with Shuttle transmitted data Perform star and horizon sensor data acquisition preparations - star selection, pointing angles, and vehicle attitude needed.	Free-Fall Navigation	Numerically integrating the state vector equations-same as Apollo. Wired logic for 3 x 3 matrix operations Integrate both own vehicles and target vehicle state vectors throughout the mission.	Digital Computer Time Reference Oscillator
	Initial State Vector Acquisition	Obtain initial states just prior to separation Whenever the target range and target state are accurately known, update own vehicle state vector. This applies when docked to Shuttle or to the Satellite, and when Laser range data is accurate.	Two-Way Data Link
	Attitude Reference Acquisition and Update	Obtain gross pitch, yaw, and roll Shuttle attitude prior to separation. Turn on attitude computation and attitude reference package prior to separation Computer integrates for body and inertial attitudes at the data rate of 1000 iterations per second at least.	Two-Way Data Link 3-axis gyro reference package, six gyros are needed to satisfy fail safe requirement.

Table 4.3-1 Proposed Guidance, Navigation and Control Concepts (Sheet 2)

GN&C EVENTS	FUNCTION	CONCEPTS PROPOSED	HARDWARE NEEDED OR AFFECTED
Separation From Shuttle Compute time to go for perigee burn (node crossing), delta vel. magnitude, direction, and duration Command RCS for star and horizon sensor data acquisition Process data and update state vector, attitude, and time to go for first burn. Count down time to go for burn initiation.	Attitude Ref. Acquisition and Update (cont.)	After separation, command own vehicle to acquire horizon sensor data while orbital rate is maintained in the vehicle pitch axis (assuming belly-mounted horizon sensors) Acquire and track a pair of stars for updating the azimuthal channel. The star trackers are driven at inertial orbital rate continuously. Both star and horizon sensor data are processed to correct vehicular and inertial attitudes continuously until update is completed.	Horizon Sensor Package Star Trackers Attitude Control and Vehicular Response
	State Vector Estimation and Correction	Once vehicular and inertial attitudes are established within expected tolerance, the horizon sensor and star data are processed to furnish position and velocity estimates. Kalman estimation technique is used to compliment free fall navigation during the collection of star and horizon sensor data. After error deviations are stabilized to established limits, free fall navigation will be solely used until the next aided navigation sequence. Aided navigation will be used during both parking and non-parking orbits as needed.	Digital computer, star trackers, and horizon sensor package

Table 4.3-1 Proposed Guidance, Navigation and Control Concepts (Sheet 3)

GN&C EVENTS	FUNCTION	CONCEPTS PROPOSED	HARDWARE NEEDED OR AFFECTED
During Perigee Burn			
Initiate burn and compute time remaining	Powered Flight Phases	For delta velocity burns, cross product steering will be used.	Thrust Vector Control and Vehicular Response
Process accelerometer data and perform navigation computations.		Acceleration due to thrust is measured and processed.	Accelerometer Package
Maintain desired vehicle attitude with proper commands to TVC and attitude control systems.			
Terminate burn when ΔV is achieved.			
After Perigee Burn			
Perform star and horizon sensor data acquisition preparations - star selection, pointing angles, and vehicle attitude needed.	Sensor Data Acquisition and Processing		
Acquire and process sensor data and update state vector and attitude.	State Vector Update		
Determine ΔV needed to correct state vector.	State Vector Error Minimization	Correct state vector with translational jet ΔV as soon as possible to minimize subsequent trajectory dispersion.	Attitude Control and Translational Jets
Command translation jets for achieving ΔV.			

Table 4.3-1 Proposed Guidance, Navigation and Control Concepts (Sheet 4)

GN&C EVENTS	FUNCTION	CONCEPTS PROPOSED	HARDWARE NEEDED OR AFFECTED
Determine time to go for midcourse and back off 1 hr for data acquisition			
Count down for data acquisition			
Continue in free fall navigation	Free Fall Navigation		
When data acquisition is impending Command for attitude acquisition.			
After attitude is acquired, start data acquisition and processing for attitude and state vactor update	Attitude Reference Update During Coast	Acquisition of horizon sensor and star data at predetermined times may be combined with state vector update sequences	
Midcourse Burn		Use translational jets for this burn.	
Perform the same functions as those during perigee burn		If ΔV is below expected error, burn will be bypassed.	
After Midcourse Burn			
Determine time to go for apogee burn.			
Determine Data acquisition parameters for 1 hr prior to apogee burn		For large burns, acquire star and horizon sensor data to update state vector and attitude prior to actual ΔV burn.	
Count down time to go for data acquisition.			

Table 4.3-1 Proposed Guidance, Navigation and Control Concepts (Sheet 5)

GN&C EVENTS	FUNCTION	CONCEPTS PROPOSED	HARDWARE NEEDED OR AFFECTED
Acquire and process data Update state vector and attitude Determine apogee burn parameters based on updated state vector Countdown for apogee burn			
During Apogee Burn Perform the same functions as those for the perigee burn			
After Apogee Burn Perform the same functions as those after the perigee burn	Data Acquisition and State Vector Correction.	Translational jets are used again to minimize error dispersion 1 hr after the main burn.	
Station Keeping and Payload Emplacement With the continuing use of star and horizon sensor data and processing, compare payload disposal positional requirements with state vector. Compute ΔV needed to meet placement objective.	Payload Emplacement	Emplacement of payload will be performed with maximum achievable accuracy to minimize subsequent corrections needed.	Attitude Control and Translational Jets.

Table 4.3-1 Proposed Guidance, Navigation and Control Concepts (Sheet 6)

GN&C EVENTS	FUNCTION	CONCEPT PROPOSED	HARDWARE NEEDED OR AFFECT
Executive ΔV correction. Command attitude control to desired vehicle attitude and release payload		Vehicle attitude will be dictated by payload's earth pointing radar requirements	
Preparation for Payload Retrieval From target and own vehicle state vectors, determine phasing orbit delta velocity parameters Acquire required attitude and executive V burn. Countdown to 70 hours and turn on rendezvous sensors Compute pointing angles for vehicle and Laser gimbals	Phasing for the 6000 n.mi. separation	72 hr phasing appears to yield acceptable delta velocity budget Either throttled down MPS or translational jets could be used for this burn Laser data will be used only during terminal phase of rendezvous and docking	Attitude Control and Translational Jets Laser transmitter and Receiver

Table 4.3-1 Proposed Guidance, Navigation and Control Concepts (Sheet 7)

GN&C EVENTS	FUNCTION	CONCEPTS PROPOSED	HARDWARE NEEDED OR AFFECTED
Terminal Rendezvous and Docking Command vehicle pointing to target line of sight based on on-board data. Command search mode until lock-on is achieved Update own vehicle state with acquired data. Activate television system when range from target is 1000 meters.	Acquisition and Processing of Laser Data	When Laser data is reliable, the target state vector will be used as the reference to correct own vehicle state vector.	
Transmit guidance and control data to Ground Control via Data Link when relative range is 400 meters. Process and execute ground commands as needed. Compute and execute braking and station keeping delta velocities as needed. Compute and execute docking maneuvers, making use of Laser data.	Remote Override mode Enabling and Processing	When data is transmitted to ground control, the on-board computer will accept over-ride signals for attitude control and translational maneuvers. Use predetermined braking gates and stand off distances. At stand off, check docking vehicle misalignment angles and residual body rates. If they are at the acceptable level, execute docking maneuvers.	**Data Link**

Table 4.3-1 Proposed Guidance, Navigation and Control Concepts (Sheet 8)

GN&C EVENTS	FUNCTION	CONCEPTS PROPOSED	HARDWARE NEEDED OR AFFECTED
Monitor docking fixture sensors Transmit these sensor data to ground control Turn off TV and sensor monitoring when commanded by ground.			
Other Mission Phases		Similar to above listed concepts will be used for other mission phases	

4.3.3 Guidance and Navigation Requirements

With the navigation concepts selected as shown in Table 4.3-1, the navigation performance accuracy can be divided grossly into two categories. First is the performance achieved during free-fall flights when star and horizon sensor data are used to provide state vector update. Second is the achievable performance during long burn, powered flight when inertial instruments are used.

The use of star and horizon sensor data for navigation has been studied and reported in Reference 4-1. Typical accuracy after 90 minutes of data processing are 3700 ft downrange, 500 ft cross track, and 500 ft altitude errors for low earth orbits. The corresponding velocity errors are 0.5, 0.5, and 4.3 ft/sec, respectively. For synchronous orbits, the maximum per channel errors after 400 minutes of data processing are 20,000 ft and 2.5 ft/sec.

During powered flight, the initial instruments are used to measure acceleration or velocity increments. The required navigation accuracy dictates the instrument performance required. With the aid of an NR error analysis program, the required instrument performance parameters were determined. These required parameters are listed in Table 4.3-2. Using these instrument requirements, the 3σ navigation errors for the six large burns are as shown in Table 4.3-3. It is apparent from these results that the guideline navigation requirements are met.

During rendezvous and docking, the Laser will provide measurement of the relative geometry. The requirements for the needed Laser are shown in Table 4.3-4. They are based on typical parameter dynamic ranges shown in Table 4.3-5.

Guidance and navigation jet propulsive requirements based on the baseline mission events are shown in Table 4.3-6. The accumulated total delta velocity is 558 ft/sec, the impulse is 369,000 lb-sec, and the accumulated burn duration is 1256 seconds. Note that the 3σ 558 fps ΔV exceeds the budgeted 490 fps by 10%. This difference is deemed acceptable because +3σ conditions are not usually encountered. A review of the individual burn durations with the proposed 4-jet, 280 lb thrust system indicates that the 233 second burn for payload retrieval is the longest. Actually, it is comprised of two burns each of which takes about 116.5 seconds. In view of the 2-minute maximum burn duration, the 280 lb thrust level is considered to be acceptable for guidance and navigation.

The rationale for the larger required ΔV's are as follows. For line item 1, each propellant settling burn typically requires 3 ft/sec based on average main propellant mass and the desired degree of settling. For line items 2, 4, 5, 6, 11, 12 and 13, the 30 ft/sec is based on the 3 sigma 15 ft/sec velocity error and an equivalent amount for position error expected. For line item 8, the ΔV is based on an allowed transit time of 72 hours and 6000 n.mi. separation between the Tug and the payload in synchronous co-orbit

Table 4.3-2. Three Sigma Navigation, Guidance, and Control Requirements

ITEMS/FUNCTIONS REQUIREMENTS	ACCURACY NASA	ACCURACY NR-SD	RANGE
Computation of Postion Computation of Velocity Computation of Attitude Computation of Attitude Rate		±0.1 ft ±0.001 fps ±0.01 deg ±0.01 deg/sec	$\pm 10^9$ ft $\pm 10^5$ fps All Attitudes 10 deg/sec
Horizon Sensors	±0.1 deg	±0.07 deg	100-20,000 nautical miles 160 deg coarse tracking
Star Tracker	±0.01 deg	±0.10 deg 0.001 Sec Electronics delay max.	120 deg cone coverage 1.5 Star Magnitude
Star Tracker/Horizon Sensor Misalign.		±15 arc-sec	Three Axes
LASER	±0.33 ft ±0.1 deg	See Table 4.3-4	See Table 4.3-4
Accelerometer (Including Temp. Effects) Bias Resolution Scale Factor Acceleration Squared Input Axis Misalignment	±0.33 ft/sec	±0.002 ft/sec^2 - - - - ±300 ppm ±10 ppm ±0.16 deg	±50 ft/sec/sec ±3.0 g's boost/re-entry (non-operating)
Gyros (Including Temp. Effects) Bias Drift Input Axis g-dependent Cross Axis g-dependent Scale Factor Input Axis Misalignment Cross Axis Misalignment	±0.1 deg/Hr ±0.3 deg/Hr/g	±0.12 deg/Hr ±0.3 deg/Hr/g ±0.3 deg/Hr/g ±300 ppm ±0.1 deg ±0.16 deg	All Attitude ±10 deg/sec

Table 4.3-3 Guidance and Navigation 3 - σ Errors at the End of Main Engine Burns

MAIN ENGINE BURNS	3- σ ERRORS IN THREE ORTHOGONAL AXES	
	VELOCITY ERROR* (fps)	POSITION ERROR ** (ft)
100 n.m. x 19323 n.m. plus 2 deg plane change	9.8 6.0 8.2 14.1 r.s.s.	4300 7000 3400 8900 r.s.s.
19323 n.m. circularization plus 26.5 deg plane change	4.2 4.9 3.4 7.2 r.s.s.	1000 4300 800 4480 r.s.s.
19323 n.m. deorbit plus 26.5 deg plane change	4.8 5.1 3.8 8.0 r.s.s.	1800 4000 800 4450 r.s.s.
270 n.m. circularization plus 2 deg plane change	7.6 5.5 6.2 11.3 r.s.s.	1000 4000 750 4120 r.s.s
270 n.m. x 100 n.m. Transfer Orbit	0.58 4.31 0.56 4.32 r.s.s.	10 3700 10 3700 r.s.s.
100 n.m. Circularization	0.58 4.31 0.56 4.32 r.s.s.	10 3700 10 3700 r.s.s.

*Velocity error guideline is 16.4 ft/sec.

**Position error guideline is 32,800 ft (low orbit) and 164,000 ft. (synchronous orbit)

Table 4.3-4 Three-Sigma Laser Requirements

PARAMETER	REQUIREMENT
Range	0 to 75 n.m. or better
Range Accuracy	± 100 ft when range is one n.m. or more ± 0.33 ft when range is 100 ft. or less ± 1.5% of range for ranges between 100 ft. and one n.m.
Range-Rate Capability	Up to 3,000 fps
Range-Rate Data and Accuracy	Optional requirement
Elevation Angular Excursion	± 45 deg.
Azimuth Angular Excursion	± 45 deg.
Angular Accuracy	± 0.1 deg.
Angular Rate Excursion	± 5 deg/sec.
Angular Rate Accuracy	± 0.01 deg/sec.
Accuracy for Roll Index	± 1.0 deg.
Roll Index Excursion	± 90 deg.

Table 4.3-5 Rendezvous Sensor Data Dynamic Ranges from 75 nmi Separation* to 500 Feet Relative Range

	Rendezvous Data Dynamic Ranges (Minima and Maxima)					
	Elevation (deg.)	El. Rate (deg/sec)	Azimuth (deg.)	Az. Rate (deg/sec)	Line of Sight (deg.) #	LOS Rate (deg/sec)
100 N.M. Orbit	-0.044 44	-1.4 0.023	-45 7.1	-0.35 0.0036	0 92	0 0.27
270 N.M. Orbit	-0.25 44	-2.5 0.020	-45 7.9	-0.48 0.0046	0 92	0 0.55
19323 N.M. Orbit	12 45	-0.28 0.0013	-0.92 0.0001	-0.0065 0.010	0 142	0 0.71

* At 75 N.M. separation,the relative trajectory contains specification three-sigma navigation errors.

Initial line of sight is defined to be the reference.

Table 4.3-6 Guidance and Navigation 3 - Translational Jet Propulsive Requirements for Baseline Mission

TRANSLATIONAL EVENTS	Required ΔV (fps)	Veh. Weight (Lb)	Required Impulse (Lb-Sec)	Burn Time # (sec)
1. Six (6) Propellant Settling Burns	18		19,000	
2. Separation From Shuttle	10	65,000	20,000	71.5
3. Correct State Vector Error 1 Hr After Injection Burn *	30	36,110	33,600	120
4. Midcourse Correction (Out-Bound)	30	36,000	33,500	120
5. State Vector Correction 1 hr. After Synchronous Oribt Circularization	30	24,100	22,500	80
6. Station Keeping	30	24,100	22,500	80
7. Deploy Payload In Orbit	10	22,000	6,820	25
8. Injection Burn for Payload Retrievel	100	21,000	65,000	233
9. Phasing and Rendezvous Burns	100	20,700	64,000	230
10. Docking Maneuvers	15	20,600	9,600	34
11. State Vector Correction 1 Hr. After Deorbit	30	15,960	14,800	53
12. Midcourse Correction (In-Bound)	30	15,900	14,800	53
13. State Vector Correction After 270 n.m. Circularization	30	9,340	8,700	31
14. Transfer Phase Initiation at 100 n.m. Orbit	50	8,930	13,900	50
15. Rendezvous Midcourse and Braking	50	8,900	13,800	50
16. Docking Manuevers	25	8,870	6,900	25
17. Accumulated Totals	588	-	369,000	1256

Based on 280 Lb thrust

* Excludes Attitude Acquisition

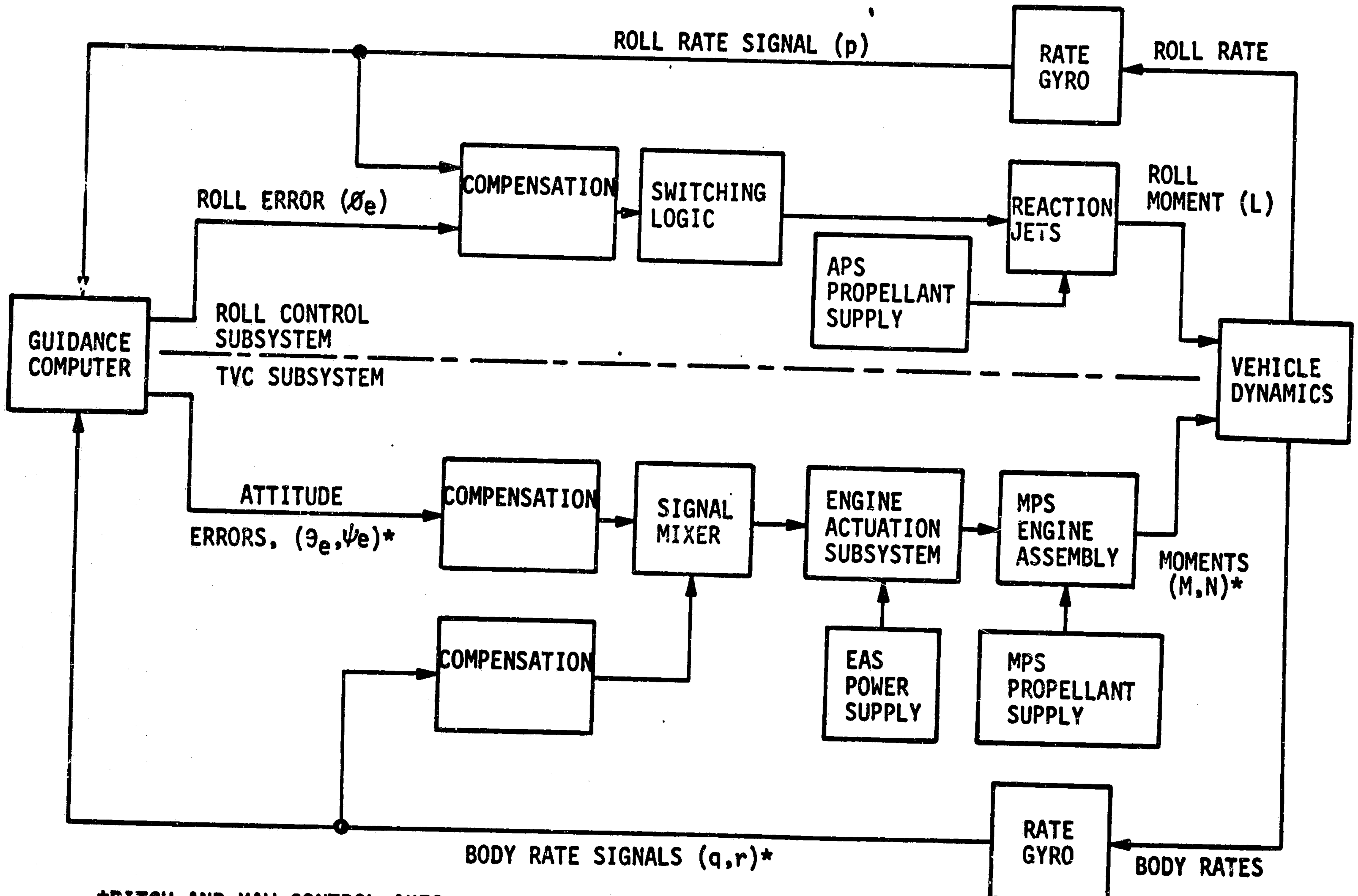
conditions. Computer program results for payload retrieval delta velocity requirements are shown in Table 4.3-7. The data shows that for 6000 n.mi. separation and 72 hr transit time, the delta velocity sensitivity is only 16 ft/sec per 1000 n.mi. For line item 9, the ΔV is based on idealized two-impulse burns needed to cancel out navigation errors when Laser data are acquired. Typical required ΔV's are shown in Table 4.3-6. To some extent the ΔV's for line items 8 and 9 are overlapping in that the braking velocity for nulling out the 6000 n.mi. separation may be combined with the rendezvous closing velocity. However, for preliminary 3-sigma estimates, the additive effect is more approximate. For line items 14 and 15, the needed ΔV's for different arrival error states and transfer phase transit angles are shown in Table 4.3-8. It is seen that the two required ΔV's needed range from 77 to 125 ft/sec, therefore 50 ft/sec for transfer phase initiation and another 50 ft/sec for midcourse and terminal rendezvous are representative requirements.

Table 4.3-9 shows the estimated main engine burns and the propulsive requirements for completing the baseline mission. The approximate burn duration ranges from 8.5 to 1300 seconds based on a 10,000 lb thrust level. It can be concluded that such a thrust level will have no impact on the guidance and navigation functions. Thrust level does, however, influence the navigation accuracy during powered flights. Based on error analysis results, the acceleration time histories for these burns do not demand inertial instruments beyond state-of-the-art. With the selected accelerometer and gyro performance parameters, the 3-σ errors for these burns are shown in Table 4.3-3. The results indicate that the NASA requirements are met.

4.3.4 TVC Requirements

The attitude control system (ACS) requirements during main propulsion system operation are shown in Table 4.3-10 for various parameters. The primary ACS requirement is thrust vector control (TVC) and fixed roll attitude. This is achieved with a gimballed main engine and reaction jets for generating roll attitude control moments. The ACS concept selected to satisfy this requirement is illustrated in Figure 4.3-2. The vehicle will have body rate stabilization in order that the ACS can operate in the fail-safe mode with guidance computer failures or loss in attitude error signals. This is achieved with a feedback control system excluding the guidance computer. Rate gyro signals feed back to the flight control computer continuously. Attitude control is provided by the outer loop closed through the guidance computer.

The ACS requirements form a basis for establishing engine actuation subsystem (EAS) requirements. Parameters for this subsystem are shown in Table 4.3-11. A comparison with MSFC guideline values shows that NR values are significantly lower except for gimbal deflection. MSFC guideline values represent state-of-the-art capability for hydraulic actuators and not Tug requirements. The differences could mean that a different type of actuation system (i.e., an electrical system) would satisfy the Tug requirements.



*PITCH AND YAW CONTROL AXES

Figure 4.3-2 Attitude Control System During Main Propulsion System Thrusting

Table 4.3-7 Delta Velocity Requirements for Synchronous, Co-Orbit Payload Retrieval (No State Vector Errors)

Separation Dist. Tug/Satellite (N.M.)	Allowed Retrieval Duration (hours)	Req'd Closing Delta Velocity (fps)	Req'd Braking Delta Velocity (fps)	Total Delta Vel. (fps)
5000	48 72 96	59 39 29	59 39 29	118 78 58
5500	48 72 96	64 43 32	64 43 32	128 86 64
6000	48 72 96	70 47 35	70 47 35	140 94 70
6500	48 72 96	76.5 51 38	76.5 51 38	153 102 76
7000	48 72 96	82 55 41	82 55 41	164 110 82

MSFC specifies 6000 N.M. separation as being maximum

Table 4.3-8 Representative Two-Impulse Rendezvous Delta Velocity Needed to Cancel State Vector Errors

	A Priori Errors Upon Arrival of Chaser			
	+Position Errors +Velocity Errors	+ Position Errors - Velocity Errors	+ Position Errors - Velocity Errors	- Position Errors - Velocity Errors
100 N.M. Circular Orbit				
Orbit Angle To Intercept, deg.	120	110	100	100
Delta Vel. For Closing, fps.	22	34	76	98
Delta Vel. For Braking, fps.	55	60	28	27
Rendezvous Delta Vel., fps.	77	94	104	125
270 N.M. Circular Orbit				
Orbit Angle To Intercept, deg.	120	110	96	100
Delta Vel. For Closing, fps	13	36	75	99
Delta Vel. For Braking, fps	25	25	26	25
Rendezvous Delta Vel., fps	43	61	101	124
19323 N.M. Circular Orbit				
Orbit Angle to Intercept, deg.	80	60	90	88
Delta Vel. For Closing, fps	75	75	90	98
Delta Vel. For Braking, fps	8	10	8	12
Rendezvous Delta Vel., fps	83	85	98	110

Assumed error magnitudes are 100,000 ft. and 8.65 fps in each orthogonal axis for the 19323 N.M. orbit, but 20,000 ft and 8.65 fps for low altitude orbits. These magnitudes are NASA requirements.

Table 4.3-9 Guidance and Navigation 3 -σ Main Engine Propulsive Requirements

Main Engine Burns	Required Delta Vel. (fps)	Vehicle Weight at Ignition (Lb)	Impulse (Lb-sec)	Burn Time (sec)*
100 n.m. x 19323 n.m. Injection (Including 2 deg plane change and gravity loss)	8615	63,832	13×10^6	13[illegible]0
19323 n.m. Circularization (Including 26.5 deg. plane change and gravity loss)	6010	35,956	5.55×10^6	555
Deorbit 19323 n.m. x 270 n.m. (Including 26.5 deg. plane change and gravity loss)	5937	23,632	3.6×10^6	360
270 n.m. Circularization (Including 2 deg plane change and gravity loss)	8025	15,888	3.07×10^6	307
270 n.m. x 100 n.m. Transfer (Including Gravity loss)	302	9,323	8.7×10^4	8.7
100 n.m. Circularization (Including gravity loss)	302	9,138	8.5×10^4	8.5
Accumulated Totals	29,200		25.4×10^6	2539

*Based on baseline 10,000 Lb thrust

Table 4.3-10 Attitude Control System Requirements During Main Propulsion System Thrusting

ATTITUDE CONTROL SYSTEM PARAMETER	REQUIREMENT BASIS	REQUIREMENT
Attitude Control Moment Generation	(1) Thrust vector control (TVC) (2) Fixed roll attitude	(1) Gimballed engine (2) Reaction jets
ACS Mode of Operation	(1) Fail safe with loss in Guidance Computer ACS commands	(1) Body rate stabilization (2) Attitude control
TVC Failure Mode	(1) Fail safe with loss in rate gyro signals (2) Minimum weight (3) System reliability	(1) APS back-up rate gyros used for MPS back-up
TVC Attitude Response Time	(1) Dynamic lag of TVC system not to affect Guidance System errors	(1) 100% step response in 2 seconds
Roll Control	(1) Roll attitude held constant (2) Roll stability with failure of one jet	(1) Regulate roll attitude (2) Four or more roll jets
Reaction Jets	(1) System reliability (2) Minimum weight	(1) APS roll jets used for MPS operation

Table 4.3-11 Attitude Control System Requirements for Engine Actuation Subsystem

ENGINE ACTUATION SYSTEM PARAMETER	ATTITUDE CONTROL SYSTEM REQUIREMENT BASIS (NAR)	ENGINE ACTUATION SUBSYSTEM PARAMETER VALUES	
		NR REQUIREMENTS	MSFC GUIDELINE
Gimbal Deflection	(1) Provide 1 deg/sec^2 vehicle control acceleration	$\pm$ 7 degrees	$\pm$ 7 degrees
	(2) Compensate for 4 inch thrust - c.g. offset		
	(3) Compensate for 1 deg thrust vector misalignment		
Gimbal Rate	(1) Sinusoidal duty cycle command at control acceleration gimbal amplitude and Attitude Control System frequency	8 deg/sec	N.S.*
Gimbal Acceleration	Same as Gimbal Rate basis	12 deg/sec^2	20 rad/sec^2 1150 deg/sec^2
Undamped Natural Frequency	(1) Dynamics of EAS and engine not to affect Attitude Control System dominant mode frequency (0.25 hz)	2.5 hz (16 rad/sec)	10 hz (630 rad/sec)
Damping factor	(1) Acceptable transient performance	0.7 (nominal)	N.S.*
	(2) Stable Attitude Control System		
Resolution	(1) Sensitivity requirement for Guidance Commands	$\pm$ 0.2 degree	N.S.*
	(2) Sensitivity requirement for correcting disturbance moments		

*Not specified

4.3.5 Flight Control Requirements for the Auxiliary Propulsion System

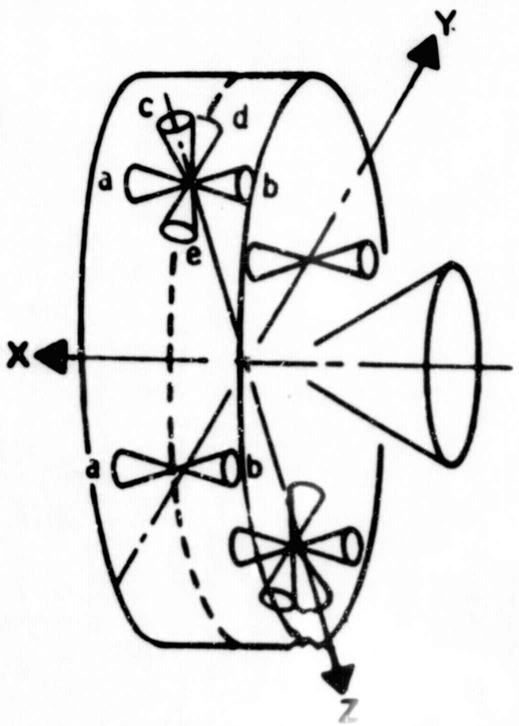
Three major design requirement areas of the Auxiliary Propulsion System (APS) are specified by vehicle flight control: thrust level, total impulse, and configuration. This section analyzes, discusses, and formulates requirements in each of the three areas.

The major requirements which dictate design and performance of the APS include a Fail Safe configuration, gaseous oxygen/hydrogen propellant, minimum weight and complexity, no violation of the EOS cargo bay envelope, and a variety of stabilization and maneuvering functions. Although the baseline mission of the Tug is to deliver and retrieve 3000-lb payloads at synchronous altitude, alternate missions to deliver an 8000-lb payload or retrieve a 4000-lb payload at synchronous altitude contain design constraints and must also be considered.

To insure adequate control capability in all operating modes, each type of operation has been analyzed separately. The thrust levels which meet all control requirements are 70 lb, with 20 lb. roll engines. The resulting bilevel system is less complex and lighter than throttlable or ganged engine approaches. This recommendation is based on current influencing criteria and should be reviewed if the criteria changes. Maximum values of impulse bit for the 70 and 20-lb engines are 2.75 and 0.65 lb-sec, respectively.

Using available mission timeline data, an analysis was conducted to determine APS total impulse requirements. The baseline mission, that of deploying a 3000-lb payload in synchronous orbit, requires 430,000 lb-sec. The 8000-lb payload delivery and 4000-lb payload retrieval missions require 184,700 and 247,300 lb-sec total impulse, respectively. Both translational and rotational operations using the RCS were included in the study.

Design problems involved with engine configuration and redundancy were attacked jointly. After a single failure of either an engine or an interfacing component, full attitude stabilization capability must be retained to permit passive docking capability and to provide safety to the EOS crew and payload. In the interest of minimizing weight and complexity, the lowest number of engines and the most centralized engine configuration were sought. The recommended system consists of 14 engines, two clusters of 5 and two clusters of 2 engines each. The arrangement is described in Figure 4.3-3, which also summarizes other APS requirements. In this part of the study, no preferred engine location along the vehicle centerline could be defined from a flight control standpoint. Quite low acceleration is provided by the APS in the lateral and vertical translation directions; and, since the vehicle center-of-mass may travel over a 16-ft span, the disturbing acceleration varies throughout the flight. Lateral and vertical thrusting also imparts a yaw or pitch torque disturbance which must be countered by other engines. The roll and axial-retro engines are canted 20-degrees to reduce plume impingement heating. Axial forward engines cannot be canted without greatly reducing their effective moment arms and thrust efficiency over long operating periods.



- **GROUND RULES**
  - FAIL SAFE ATTITUDE CONTROL
  - MINIMUM WEIGHT
  - MIDCOURSE CORRECTION CAPABILITY
  - GO_2/GH_2 PROPELLANT (I_{SP} = 380 SEC)

- **ACCURACY REQUIREMENTS FOR DOCKING**
  - CENTERLINE MISS DISTANCE: 0 TO 1.0 FT
  - MISS ANGLE: 0 TO 5.0 DEG
  - LONGITUDINAL VELOCITY: 0.1 TO 1.0 FT/SEC
  - LATERAL VELOCITY: 0 TO 0.3 FT/SEC
  - ANGULAR VELOCITY: 0 TO 0.5 DEG/SEC

- **DESIGN REQUIREMENTS**
  - 14 ENGINES
  - 430,000 LB-SEC TOTAL IMPULSE
  - 25 MS MIN SQUARE PULSE DURATION
  - NOMINAL ENGINE STATION ≅ 146.5 IN.

	ENGINE			
	a	b	c	d,e
• THRUST (LB)	70	70	70	20
• CANT ANGLE (DEG)	20	0	0	20
• MAX PULSE (SEC)	120	360	30	60

Figure 4.3-3 Auxiliary Propulsion System Flight Control Description

All of the factors leading to the conclusions are described and discussed in the next three sections.

Thrust Level Requirements

Engine thrust level is indicated by a variety of requirements which may be met by several methods, the most common of which are throttling, selective multiple engines, and bilevel clusters of engines. These methods, and the driving requirements, are discussed in the following paragraphs.

High Thrust Drivers

There are two types of spacecraft operations which require large APS thrust. These are the midcourse velocity corrections which are too small for main engine operation, and the large attitude maneuvers. Both are constrained by time limitations which are difficult to identify.

APS Midcourse Corrections

To formulate APS midcourse correction requirements, a number of assumptions are necessary. The main engine has the capability to produce a total minimum impulse of 21,000 lb-sec efficiently. Midcourse corrections which require less than 21,000 lb-sec will be accomplished by the APS. Assuming that 4 minutes is a reasonable maximum time for the correction, the total APS thrust in the forward longitudinal direction should be greater than 90 lb. If 2 engines are used, the minimum thrust per engine is approximately 45-lb to assure main propulsion overlap. Thrust is shown parametrically in Figure 4.3-4.

APS Attitude Maneuvers

Large attitude changes are frequently necessary to align the vehicle prior to main propulsion ignition, to orient for midcourse correction burns by the APS, and to point sensors. These maneuvers should ordinarily be accomplished quickly; however, propellant usage increases with diminishing maneuver time and should also be considered. Figure 4.3-5 shows parametric data for an end-for-end maneuver, performed in one minute, for various fuel efficiencies. The expected range of moment-of-inertias is 50,000 to 250,000 slug-ft^2 during this maneuver. It is evident from the figure that adequate fuel efficiency is obtained over this range for engines of approximately 70 lb.

Low Thrust Drivers

The APS must provide the precision maneuver capability dictated by docking requirements. These requirements are shown for the Tug in Table 4.3-12 and are compared with those of Apollo and OOS. The Tug requirements will be used to derive maximum APS thrust levels for both Tug/payload and EOS/Tug docking operations.

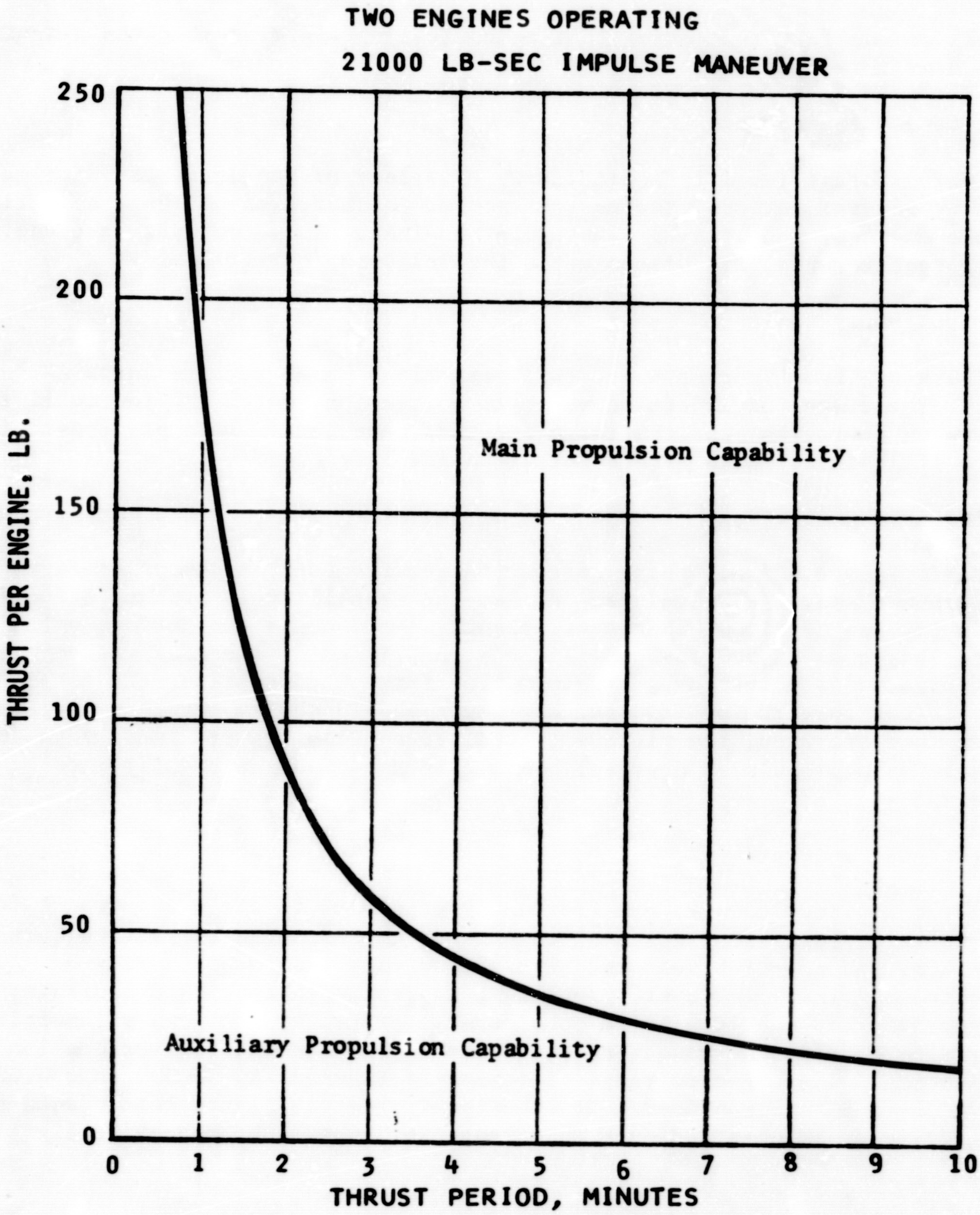


Figure 4.3-4 APS Midcourse Correction Capability

- TWO ENGINES OPERATING
- 10-DEGREE CANT ANGLE
  (MOMENT COUPLE WITH 0^{o} & 20^{o} CONTROL ENGINES)
- 180-DEGREE MANEUVER
- ONE-MINUTE MANEUVER TIME
- 7-FT MOMENT ARM

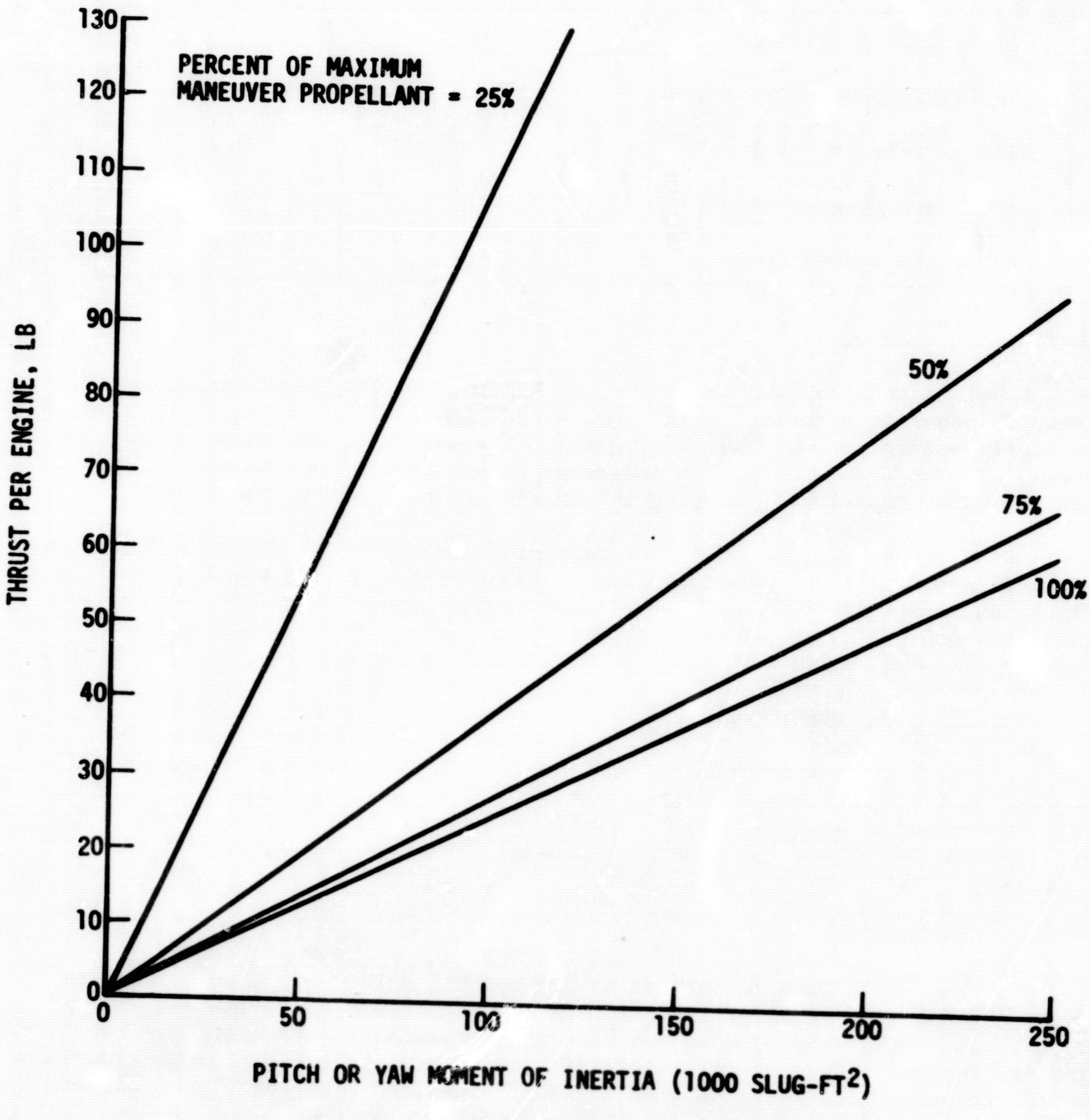


Figure 4.3-5 APS Attitude Maneuver Capability

Table 4.3-12. Comparison of Active Vehicle Docking Accuracy Specifications

Variable	Apollo and OOS Requirements	Tug Point Design Structural Requirements
Miss Distance (Inches)	± 12	± 12
Longitudinal Velocity (Ft/Sec)	0.1 to 1.0	0.1 to 1.0
Lateral Velocity (Ft/Sec)	0.5	0.3
Miss Angle (Deg)	± 10	± 5
Angular Rate (Deg/Sec)	1.0	0.5

Tug/Payload Docking

Previous studies have shown that an APS thrust level which produces at least 10 separate identical change maneuvers within the requirement envelope is easily controllable. The most constraining envelopes are those of angular rates. In the case of Tug, this envelope is one degree/sec; therefore, the APS should be capable of changing the angular rate by 0.1 degrees/sec.

Assuming two engines, a 7-ft moment arm, and a minimum APS pulse duration of 0.025 seconds, torque calculations reveal that the thrust level is 0.00506 times the moment-of-inertia. These thrust values are plotted parametrically in Figure 4.3-6. As is always the case with a long, cylindrically-shaped vehicle, the roll moment of inertia is much smaller than that of pitch or yaw.

The pitch or yaw angular velocity bound permits thrust levels which are of the same magnitude (in the region of 70-lb per engine) as was found by large APS thrust considerations. Roll thrust, as bounded by the maximum roll angular rate constraint, is approximately 20-lb at the predicted nominal roll moment-of-inertia value of 5,000 slug-ft^2. Note that the thrust levels are well within the bounds and thus permit small variations in all contributing factors.

EOS/TUG Docking

The Tug, as a passive partner in the docking role, must observe requirements for the same level of precision as the active vehicle. These requirements are assumed to be the same as the stated requirements for Tug/payload docking. Under these conditions the APS design for active docking will also be applicable to passive docking. Thus, no additional requirements are imposed on APS design due to EOS/Tug docking.

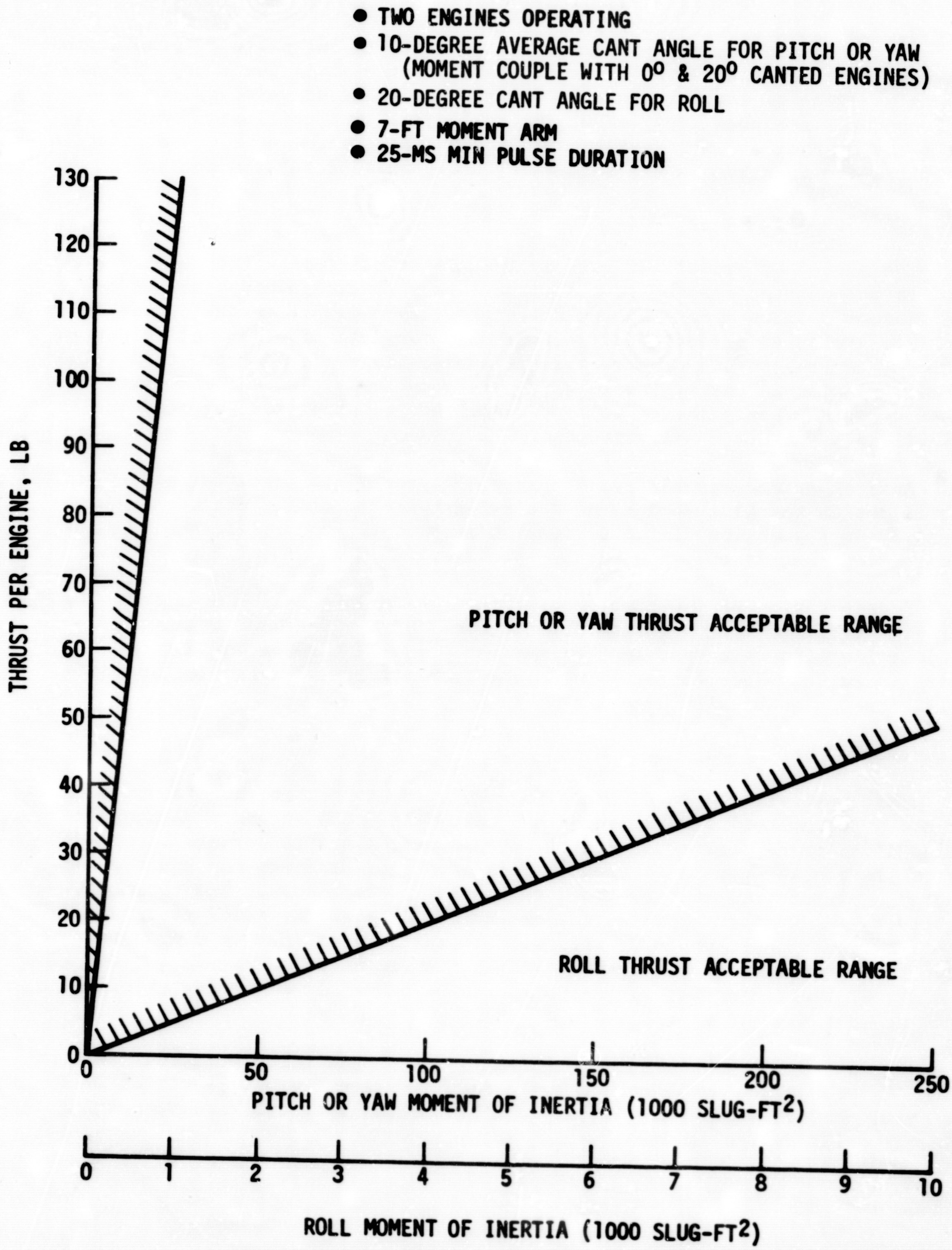


Figure 4.3-6 APS Capability for Tug/Payload Docking

APS Roll Control for Main Propulsion Operation

A single main engine, gimballed in two axes, is incapable of stabilizing the Tug about the roll axis. Thus, the APS provides this function. Unless the main engine exhaust has a large "swirl" component of velocity, no major prolonged disturbances occur about the roll axis. The conclusion, which is borne out by Apollo flight data, is that the APS performs a normal attitude stabilization function in the roll axis during main propulsion operation.

Attitude Stabilization Propellant Efficiency

The amount of propellant used by the APS to maintain attitude stabilization throughout the mission is minimized in a well designed system. Ideally, the minimum results from a system which provides adequate torque authority over all prolonged periodic and secular disturbances. The most efficient ratio of control-to-disturbance torques appears to be 7; however, this value usually results in thrust levels which are much lower than those derived from other drivers. Hence the propellant usage rate to stabilize Tug attitude is established by other considerations.

An appropriate equation which describes propellant usage for attitude stabilization was derived and checked against Apollo flight data. A major simplifying assumption of the derivation is that the control torque is much greater than all disturbance torques present.

A description of the equation is given in Table 4.3-13. Note that the Tug structural docking requirements are used for the deadband amplitude. Minimum inertias were used so that maximum propellant rates would result. The calculation shows that the rate is approximately 1.2 lb/hr for attitude stabilization about all three axes.

Thrust Level Selection

This analysis was conducted using the assumption that the minimum equivalent square wave pulse duration of APS thrust is 0.025 seconds. A shorter pulse duration capability is desirable, however, since it permits larger thrust levels to be used in translations and major attitude maneuvers without degrading vernier capability for docking and attitude hold. The capability to develop much shorter pulses is now indicated by engine manufacturers; thus, the APS requirement should be specified in terms of minimum impulse rather than thrust level. Appropriate relationships are shown in Figure 4.3-7. A summary of vehicle geometry, mass properties, and minimum impulse critical to docking operation is given in Table 4.3-14.

Recent engine manufacturer studies indicate that pulse durations down to 25 milliseconds are achievable. Based on this value, thrust levels of 20 lb per roll control engine and 70 lb for all other engines are recommended for the Tug point design study.

Table 4.3-13. Attitude Stabilization Propellant Usage Estimate

Approximate Equation

$$\dot{W}_p = \frac{57.3 \times 3600 \, (nF \, \Delta t_{min})^2 \, d}{8I \, Isp \, \theta_{db}} \qquad \text{Lb/Hr}$$

Nomenclature and values

$nF = 70(1 + \cos 20°)$, pitch or yaw thrust force, lb

$nF = 20 \; (2 \cos 20°)$, roll thrust force, lb

$\Delta t_{min} = 0.025$, Minimum equivalent square wave pulse duration, seconds

$d = 14$, Opposing engine cluster diameter, ft

$I_{sp} = 0.90 \times 415$, Effective specific impulse, sec

$\theta_{db} = 1.0$, Attitude deadband amplitude, degrees

I = Moment of inertia about the vehicle cg, slug-ft^2

Rate Computation

$$W_p = 0.605 \, \frac{(nF)^2}{I}$$

Axis	Thrust Per Engine (Lb)	Moment of Inertia (Slug-ft^2)	Rate (lb/hr)
Roll	20	5180	0.164
Pitch	70	21678	0.510
Yaw	70	21678	0.510
Total			1.184

- O TUG POINT DESIGN DOCKING PARAMETERS
- O TWO ENGINES OPERATING
- O 7-FT MOMENT ARM
- O ROLL MOMENT OF INERTIA = 5180 SLUG-FT^2
- O PITCH OR YAW MOMENT OF INERTIA = 21678 SLUG-FT^2

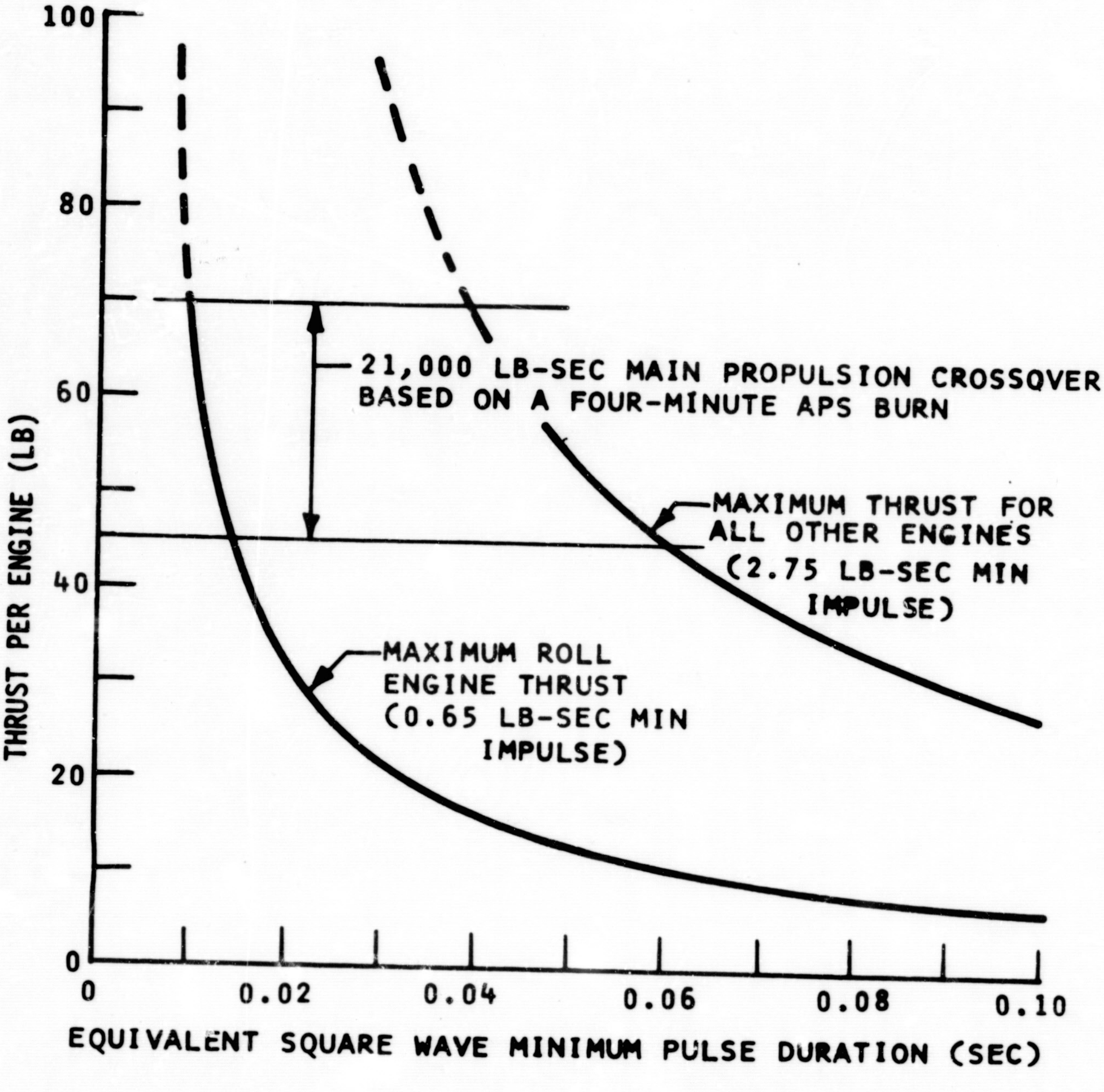


Figure 4.3-7 Variation of Recommended Thrust Levels with Minimum Pulse Duration

Table 4.3-14 Summary of Tug Mass Properties and APS Thrust Levels*

Parameter	Mission		
	Baseline	8000 Lb Delivery	4000 Lb Retrieval
Engine Moment Arm (Ft)	7	7	7
Payload Docking Phase			
Weight (lb)	21000	--	22940
Pitch or Yaw Moment-of-Inertia (Slug-Ft^2)	42000	--	44563
Roll Moment of Inertia (Slug-Ft^2)	$\boxed{5180}$	--	$\boxed{5180}$
EOS Docking Phase			
Weight (lb)	8890	5890	10050
Pitch or Yaw Moment-of-Inertia (Slug-Ft^2)	77166	$\boxed{21678}$	90700
Roll Moment of Inertia (Slug-Ft^2)	7799	$\boxed{5180}$	8812
Thrust Levels			
Roll Engines (lb)	20	20	20
All Other Engines (lb)	70**	70	70**

*Weight and moment of inertia data based on 11-17-71 Mass Properties Estimates

$\boxed{}$ Indicates critical parameters

**Maximum thrust set by midcourse correction criteria, all others set by docking requirements

Total Impulse Requirements

Impulse requirements were estimated for three Tug missions which include the following modes of operation:

(1) Attitude hold (stabilization)

(2) Attitude maneuvers

(3) Propellant settling

(4) Translation maneuvers

The impulse requirements for each of these modes are discussed in the next paragraphs, and are followed by a summary. The total impulse design requirement is 430,000 lb-sec.

APS Timeline of Events

Each mission was dissected into elements appropriate for impulse analysis. These timelines appear in Tables 4.3-15 and 4.3-16.

Attitude Stabilization

During attitude hold or local vertical hold the vehicle will limit cycle about the roll, pitch and yaw axes.

The impulse required for limit cycle operation in each axis is given by

$$I_{A.H.} = \frac{57.3\ (nF\ \Delta t_{min})^2\ d\ \Delta t}{8\ \theta_{db}\ I}$$

where

$I_{A.H.}$ = impulse for attitude hold, lb-sec

n = number of operating jets

F = APS thrust, lb

Δt_{min} = impulse bit duration, sec

d = stage diameter, ft

θ_{db} = attitude deadband, deg

I = moment of inertia, slug-ft^2

t = attitude hold mode duration, sec

Table 4.3-15 Auxiliary Propulsion System Timeline of Events (3000 lb Payload Mission)

Mission Phase	Event Description	Phase Duration (HMS)	Auxiliary Propulsion System Modes: Attitude Maneuver* (deg)	Hold Attitude Duration* (HMS)	Settle Propellant Mode?	ΔV Budget (ft/sec)
1	Separation of Tug from Shuttle	2.0.0	1-180°P, 1-90°R	2.0.0 (R, Y, P)	No	10
2	In-orbit coast for proper phasing	12.0.0	1-180°P, 1-90°R	12.0.0 (R, Y, P)	No	0
3	2° plane change and 100 x 1930 n mi transfer orbit burn	0.20.40 **	1-180°P, 1-90°R	0.20.40 (R)	Yes	0
4	Coast to 1930 n mi apogee; mid-course correction	5.24.0	1-180°P, 1-90°R	5.24.0 (R, Y, P)	No	50
5	2 6.5° plane change and circularization burn	0.9.30 **	1-180°P 1-90°R	0.9.30 (R)	Yes	0
6	Station keeping, deploy payload	1.0.0	1-180°P, 1-90°R	1.0.0 (R, Y, P)	No	40
7	In-orbit phasing for payload retrieval	72.0.0	1-180°P, 1-90°R	72.0.0 (R, Y, P)	No	100
8	Perform rendezvous with payload	2.0.0	1-180°P, 1-90°R	2.0.0 (R, Y, P)	No	100
9	Dock with payload	1.0.0	1-180°P, 1-90°R	1.0.0 (R, Y, P)	No	15

*Pitch (P), Yaw (Y), and Roll (R) axes designated

**MPS burn duration only. Additional time is necessary to orient vehicle and settle propellants prior to each burn.

Table 4.3-15 Auxiliary Propulsion System Timeline of Events (Cont'd)
(3000 lb Payload Mission)

Mission Phase	Event Description	Phase Duration (HMS)	Auxiliary Propulsion System Modes: Attitude Maneuver* (deg)	Hold Attitude Duration* (HMS)	Settle Propellant Mode?	ΔV Budget (ft/sec)
10	Perform payload safing and return trip phasing	12.30.0	1-180°P, 1-90°	12.30.0 (R, Y, P)	No	0
11	26.5° plane change and 19,300 x 270 n mi injection burn	0.6.0 **	1-180°P, 1-90°R	0.6.0 (R)	Yes	0
12	Coast to 270 n mi orbit, mid-course correction	5.20.0	1-180°P, 1-90°R	5.20.0 (R, Y, P)	No	50
13	2° plane change and 270 n mi circularization burn	0.5.10 **	1-180°P, 1-90°R	0.5.10 (R)	Yes	0
14	Wait for proper phasing	22.0.0	1-180°P, 1-90°R	22.0.0 (R, Y, P)	No	0
15	Injection burn for 270 x 100 n mi orbit	0.16.54 **	1-180°P, 1-90°R	0.16.54 (R)	Yes	0
16	Coast to 100 n mi perigee	0.45.36	1-180°P, 1-90°R	0.45.36 (R, Y, P)	No	0
17	Circularize 100 n mi orbit	0.0.14 **	1-180°P 1-90°R	0.0.14 (R)	Yes	0
18	Rendezvous with Shuttle	1.0.0	1-180°P 1-90°R	1.0.0 (R, Y, P)	No	100
19	Dock with Shuttle	1.0.0	1-180°P, 1-90°R	1.0.0 (R, Y, P)	No	25

Table 4.3-16 Auxiliary Propulsion System Timeline of Events
(8,060 lb Payload Mission or 4,160 lb Payload Mission)

Baseline Mission Phase Number	Event Description	Phase Duration (HMS)	Auxiliary Propulsion System Modes			
			Attitude Maneuver* (deg)	Hold Attitude Duration* (HMS)	Settle Propellant Mode?	ΔV Budget (ft/sec)
1	Separation from Shuttle	2.0.0	1-180°P, 1-90°R	2.0.0 (R, Y, P)	No	10
2	In-orbit coast for proper phasing	12.0.0	1-180°P, 1-90°R	12.0.0 (R, Y, P)	No	0
3	2° plane change and 100 x 1930 n mi transfer orbit burn	**	1-180°P, 1-90°R	(R)	Yes	0
4	Coast to 1930 n mi apogee mid-course correction	5.24.0	1-180°P, 1-90°R	5.24.0 (R, Y, P)	No	0
5	26.5° plane change and circularization burn	0.9.30 **	1-180°P, 1-90°R	0.9.30 (R)	Yes	0
6 or 8, 9	Deploy 8,060 lb payload, station keeping Rendezvous and dock with 4,160 lb P.L., stationkeeping	1.0.0	1-180°P 1-90°R	1.0.0 (R, Y, P)	No	55
10	Return trip phasing	18.0.0	1-180°P, 1-90°R	18.0.0 (R, Y, P)	No	0
11	26.5° plane change and 19,300 x 270 n mi injection burn	**	1-180°P 1-90°R	(R)	Yes	0

*Pitch (P), Yaw (Y), and Roll (R) axes are indicated

**MPS burn duration only. Additional time is necessary to orient vehicle and settle propellants prior to each burn.

Table 4.3-16 Auxiliary Propulsion System Timeline of Events (Cont'd)
(8,060 lb Payload Mission or 4160 lb Payload Mission)

Baseline Mission Phase Number	Event Description	Phase Duration (HMS)	Auxiliary Propulsion System Modes			
			Attitude Maneuver* (deg)	Hold Attitude Duration* (HMS)	Settle Propellant Mode?	ΔV Budget (ft/sec)
12	Coast to 270 n mi orbit, mid-course correction	5.20.0	1-180°P, 1-90°R	(R, Y, P)	No	50
13	2° plane change and 270 n mi circularization burn	0.5.10 **	1-180°P, 1-90°R	(R)	Yes	0
14	Wait for proper phasing	22.0.0	1-180°P, 1-90°R	(R, Y, P)	No	0
15	Injection burn for 270 x 100 n mi orbit	0.16.54 **	1-180°P, 1-90°R	(R)	Yes	0
16	Coast to 100 n mi orbit	0.45.36	1-180°P, 1-90°R	(R, Y, P)	No	0
17	Circularize 100 n mi	0.0.14 **	1-180°P, 1-90°R	(R)	Yes	0
18	Rendezvous with Shuttle	1.0.0	1-180°P 1-90°R	(R, Y, P)	No	100
19	Dock with Shuttle	1.0.0	1-180°P 1-90°R	(R, Y, P)	No	25

Given $n = 2$, $t_{min} = 0.05$ sec*, $d = 13.4$ ft, $t = 1$ hr, $\theta_{db} = 5.0$ deg, $F = 20$, 70 lb for roll, and pitch axes, respectively. Then the impulse per hour is

$$I_{AH} \left\{ \begin{array}{l} 2.76 \times 10^5/I_x, \text{ roll axis} \\ 3.386 \times 10^6/I_y, \text{ pitch axis} \\ 3.386 \times 10^6/I_z, \text{ yaw axis} \end{array} \right.$$

The timelines for hold attitude mode for the three axes are shown in Tables 4.3-15 and 4.3-16 for the 3,000 lb payloads and 8,060 lb or 4,160 lb payloads respectively. Based on these timelines and Tug inertias the impulse required during performance of the three missions for attitude hold are shown in Table 4.3-17. The total impulse required is 35,100, 20,100, and 5,000 for the 3,000 lb, 8,060 lb, and 4,160 lb payload missions respectively.

Attitude Maneuvers

Figure 4.3-8 shows a typical attitude maneuver about a single axis. It is characterized by acceleration, coast, and deceleration phases. The acceleration impulse is required to achieve the constant rate of $\theta_{max} I/d$,

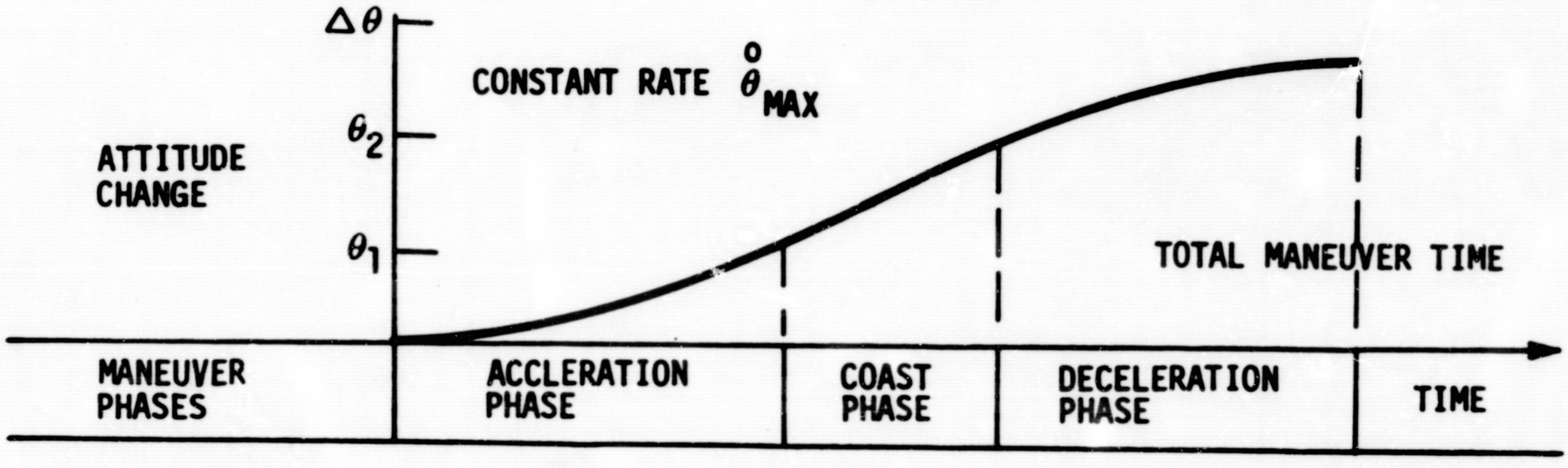


Figure 4.3-8 Attitude Maneuver Mode

*Selected for establishing impulse requirements

Table 4.3-17 Auxiliary Propulsion System Impulse Required For Attitude Hold Mode & For Three Missions

Baseline Mission Phase Number*	Auxiliary Propulsion System Impulse Required for Attitude Hold Mode Per Mission ***		
	3,000 Lb Payload Mission	8,060 Lb Payload Mission	4,160 Lb Payload Mission
1	168 lb-sec	100 lb-sec	310 lb-sec
2	1004	640	1840
3	17	10	30
4	502	300	760
5	10	10	15
6	265	340	N.A.**
7	19,296	N.A.**	N.A.
8	536	N.A.	40
9	269	N.A.	40
10	8,850	6,030	1540
11	4	10	10
12	735	1780	10
13	4	10	10
14	3052	9530	
15	9	20	10
16	111	350	100
17	0	0	0
18	139	470	120
19	139	470	120
Totals (lb-sec)	35,114	20,070	4,955

*See Table 4 for phase definitions
**Not Applicable
***Includes 20% contingency

where 'I' is the inertia and 'r' is the stage diameter. This same amount of impulse is required to decelerate the vehicle to zero rate so the total impulse required to complete the attitude maneuver is

$$I_{A.M} = \frac{2\theta_{max} I}{r}$$

A value of 2 deg/sec is assumed for the maximum maneuver rate, based on 0.01 deg/sec resolution required for the rate gyros and a maximum-to-minimum rate range of 200:1. Complete spherical attitude coverage for the vehicle can be obtained by a 90° maximum roll followed by a 180° maximum pitch or yaw maneuver of the type described above. The design assumes complete spherical coverage capability for every phase of each mission. This is indicated in the attitude maneuver mode columns of Table 4.3-15 and 4.3-16. The impulse required for each mission phase is shown in Table 4.3-18. The impulse design requirement including 20% contingency is 6300 lb-sec. This is necessary to satisfy the requirements for the 8060-lb payload delivery mission. This mission has the largest inertias and thus requires more propellant for the same maneuver rate. For the 3,000 and 4,160 lb payload missions the impulse requirements are 6,180 and 4,550 lb-sec respectively.

Propellant Settling

Prior to each main propulsion burn the propellants must be settled. The necessary axial acceleration will be provided by the APS. In this case the total impulse relationship is

$$I_{P.S} = 17.2 \sqrt{W}$$

where

$$W = \text{stage weight, lb}$$

Impulse details and total for each mission and for each MPS burn are shown in Table 4.3-19. The total impulse (including 20% contingency) is 18,950, 17,270 and 14,916 lb-sec for the 3,000, 8,060, and 4,160 lb payload missions respectively.

Translation Maneuvers

Translation maneuvers of less than 100-ft/sec will be performed by the APS. These include midcourse correction, and operations involved in rendezvous and docking. The ΔV budget for the three missions is shown in Table 4.3-20 for each phase. The impulse required for a translation maneuver, I_{TM}, can be determined from

$$I_{TM} = M \Delta V \quad \text{(lb-sec)}$$

Table 4.3-18 Auxiliary Propulsion System Impulse Required For Attitude Maneuver Mode (Includes 20% Distributed Contingency)

Baseline Mission Phase Number*	Auxiliary Propulsion System Impulse Required for Attitude Maneuver Mode		
	3,000 Lb Payload Mission	8,060 Lb Payload Mission	4,160 Lb Payload Mission
1	520 lb-sec	1,010 lb-sec	84 lb-sec
2	510	1,010	84
3	430	905	70
4	430	875	70
5	396	820	70
6	396	820	N.A.
7	396	N.A.**	N.A.
8	396	N.A.	500
9	396	N.A.	500
10	396	100	500
11	330	100	500
12	320	100	500
13	220	80	310
14	208	80	310
15	208	80	310
16	208	80	310
17	208	80	310
18	208	80	310
19	208	80	310
Totals	6,176 lb-sec	6,300 lb-sec	4,550 lb-sec

*See Table 4 for phase definitions
**Not applicable

Table 4.3-19 Auxiliary Propulsion System Impulse Required to Settle Propellants (Includes 20% Distributed Contingency)

Baseline Mission Phase Number*	Auxiliary Propulsion System Impulse Required To Settle Propellants (lb-sec)		
	Mission Payload (Lb)		
	3,000	8,060	4,160
1	N.R.**		
2	N.R.		
3	5,260	5,270	5,160
4	N.R.		
5	3,930	3,960	3,840
6	N.R.		
7	N.R.		
8	N.R.		
9	N.R.		
10	N.R.		
11	3,190	2,640	3,480
12	N.R.		
13	2,610	2,160	2,760
14	N.R.		
15	2,000	1,680	2,160
16	N.R.		
17	1,960	1,560	2,160
18	N.R.		
19	N.R.		
Totals (Lb-sec)	18,950	17,270	19,560

*See Table 4 for phase definitions
**Not required

Table 4.3-20 Auxiliary Propulsion System Impulse Required For Translation Maneuver Mode (Includes 20% Distributed Contingency)

Baseline Mission Phase Number*	Translation Budget ΔV, (ft/sec)			Auxiliary Propulsion System Impulse Required for Trans. Mode (Lb-Sec)		
	Mission Payload (Lb)			Mission Payload (Lb)		
	3,000	8,060	4,160	3,000	8,060	4,160
1	10	10	10	23,900	24,200	23,160
2	N.R.**					
3	N.R.					
4	50	50	50	67,300	68,160	56,280
5	N.R.					
6	40	55	N.A.	49,000		N.A.
7	100	N.A.***	N.A.	65,530	N.A.	N.A.
8	100	N.A.	40	89,070	N.A.	40,930
9	15	N.A.	15	11,700	N.A.	15,350
10	N.R.					
11	N.R.					
12	50	50	50	24,830	20,300	34,320
13	N.R.					
14	N.R.					
15	N.R.					
16	N.R.					
17	N.R.					
18	100	100	100	27,860	22,730	38,390
19	25	25	25	8,300	5,640	9,720
Totals (lb-sec)				367,500	141,030	218,150

*See Table 4 for phase definitions
**Not required
***Not applicable

where M is the stage mass (slug), and ΔV is change in velocity (ft/sec). The impulse required for each phase for the missions are shown in the last three columns of Table 4.3-20. The total impulse requirements are 367,000, 141,030, and 218,130 lb-sec for the 3,000 lb, 8,060 lb, and 4,160 lb payload mission, respectively.

Total Impulse Summary

The complete requirements for APS total impulse are given in Table 4.3-21. These reflect all identifiable operations to be operformed by each of the three missions.

Engine Configuration and Redundancy Requirements

The APS must be capable of controlling three axes of rotation and translation. Under fail-safe conditions the APS must retain three axis rotation control in order to be retrieved, and also must be capable of longitudinal translation control to prevent hazards to the EOS crew and the Tug payload. The plumbing and valving design must preclude a thrust-on type of failure and a plumbing leak.

Engine Redundancy

An analysis of an Apollo configuration, using 16 engines in four identical clusters, is reproduced in Table 4.3-22. The first part of the table considers the case where the engine station is too near the vehicle c.g., to allow use of the longitudinal moment arm (say, within approximately two feet). The lower part describes conditions further from the vehicle cg. Comparison of the first and second parts of the table show that there is no apparent advantage to locating the APS in a special position relative to the vehicle cg if only one engine failure is considered. The 16-engine system allows the failure of any single engine without the loss of either rotational or longitudinal translational control.

An analysis of a 14-engine system is shown in Table 4.3-23. The configuration consists of two clusters of five engines (pentads) and two clusters of two engines (duads). The pentads are oriented such that one engine is radial, two are opposing and point in the circumferential direction, and two are opposing and point in the axial direction. The duads are oriented so that the two engine oppose and point in the axial direction. The capability of this system is similar to that of the Apollo system until one jet fails, whereupon lateral translational capability is lost.

The conclusion is that either the 16- or 14-engine configuration meets the requirements of fail safe operation after a single engine failure. An added restriction is that the engine fails with zero thrust.

Table 4.3-21 Auxiliary Propulsion System Impulse Requirements

Auxiliary Propulsion System Mode	Impulse Required Per Mission *(Lb-Sec) Mission Payload (Lb)		
	3,000	8,060	4,160
Hold Attitude Mode	35,100	20,100	5,000
Attitude Maneuver Mode	6,176	6,300	4,550
Propellant Settling Mode	18,950	17,270	19,560
Translation Maneuver (ΔV) Mode	367,500	141,030	218,150
Totals (Lb-sec)	427,726	184,700	247,260

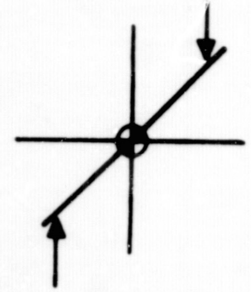
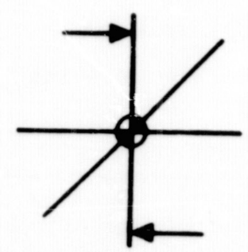
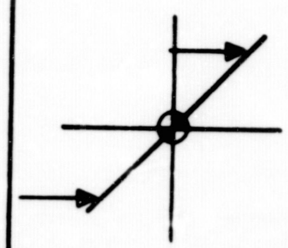
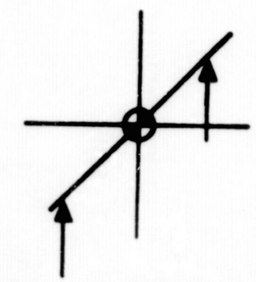
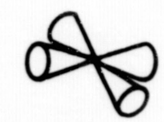
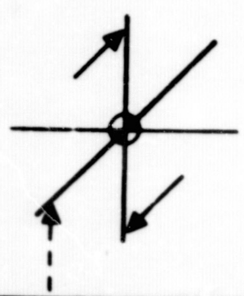
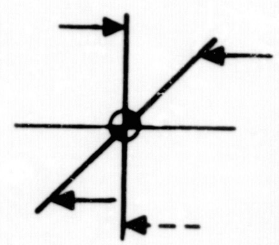
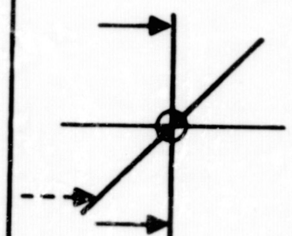
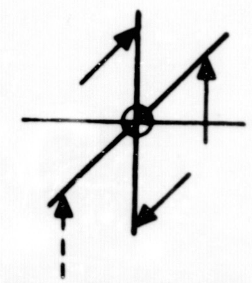
*Includes 20% contingency

Table 4.3-22 APS Redundancy Analysis for a 16-Engine System

→ INDICATES JETS USED

--→ INDICATES JETS FAILED

• ENGINE STATION NEAR VEHICLE CG

FAILURE LEVEL	ROLL	PITCH OR YAW	LONG TRANS	LAT OR VERT TRANSLATION	COMMENTS
NORMAL OPERATION					 • OPERATIONAL
FIRST FAILURE					• LOGIC CHANGE IN ROLL • LOGIC CHANGE IN • LONGITUDINAL TRANSLATION

• ENGINE STATION NOT NEAR VEHICLE CG

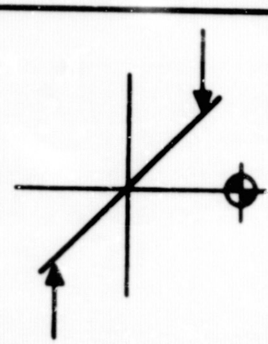
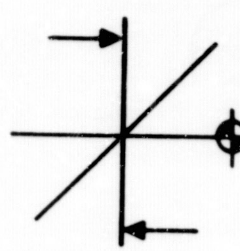
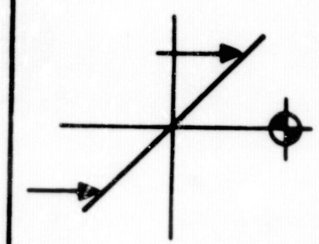
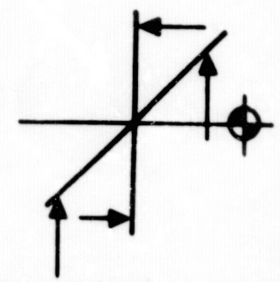
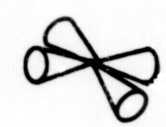
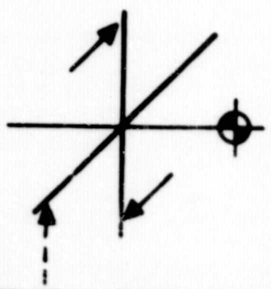
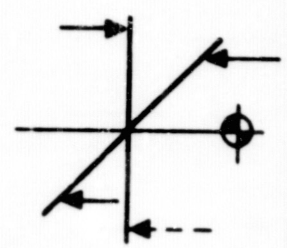
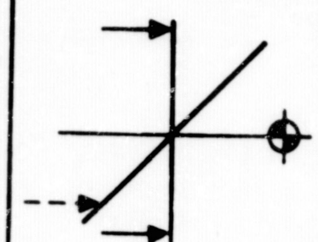
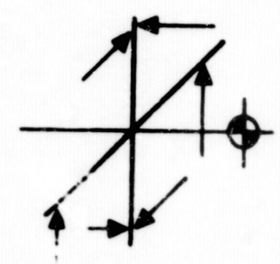
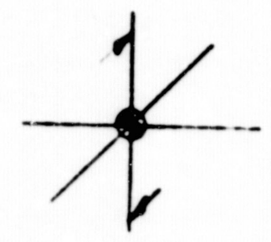
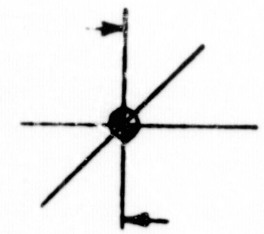
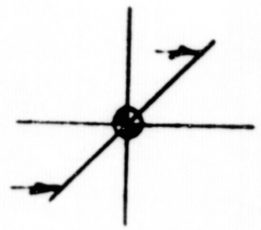
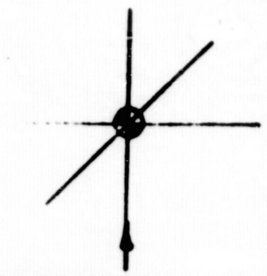
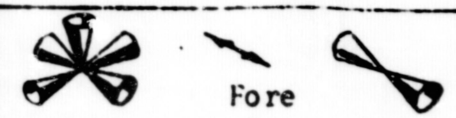
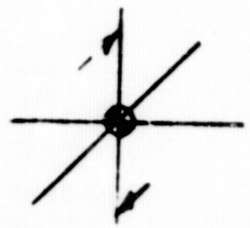
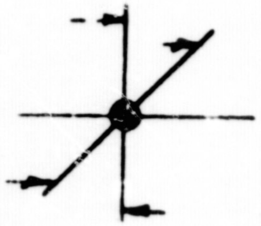
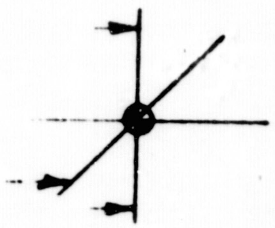
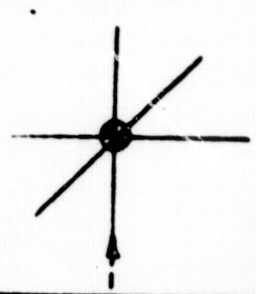
FAILURE LEVEL	ROLL	PITCH OR YAW	LONG TRANS	LAT OR VERT TRANSLATION	COMMENTS
NORMAL OPERATION					 • OPERATIONAL
FIRST FAILURE					• LOGIC CHANGE IN ROLL • LOGIC CHANGE IN • LONGITUDINAL TRANSLATION

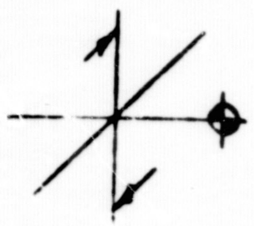
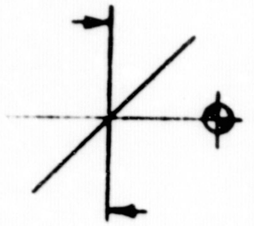
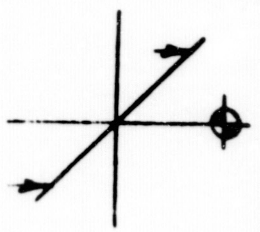
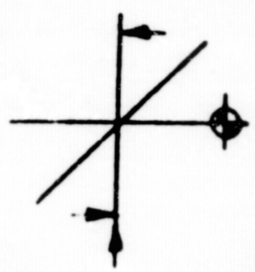
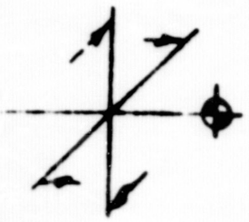
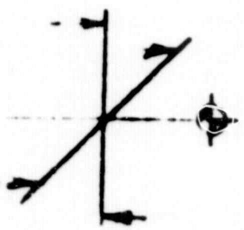
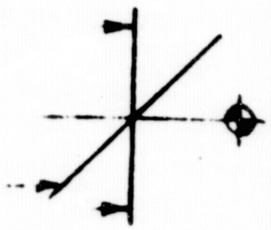
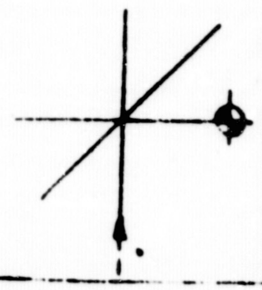
Table 4.3-23 APS Redundancy Analysis for a 14-Engine System

Engines Used
Engines Failed

o Engine Station Near Vehicle CG

Failure Level	Roll	Pitch or Yaw	Long Trans	Lat or Vert Translation	Comments
Normal Operation					 Top and Bottom; Fore & Aft; Sides
Single Failure					o Lateral Trans From Roll o Logic Change for Long Trans o Lateral Trans Disabled

o Engine Station Not Near Vehicle CG

Failure Level	Roll	Pitch or Yaw	Long Trans	Lat or Vert Translation	Comments
Normal Operation					o Operational
Single Failure					o Lateral Trans from Roll o Logic Change for Long Trans o Lateral Trans Disabled

Engine Cant Angle Considerations

All of the engines except those which produce forward axial acceleration and those which are radially oriented, may be canted through small angles without loss of control. These engines are not operated for prolonged periods, hence the resulting propellant loss is not excessive. If the engines are mounted on the aft cylindrical section of the vehicle, the vehicle center of mass is 20 ft forward of the engine station in some cases. Even a 5-degree cant angle with the forward axial acceleration engines greatly reduces the effective moment arm under these conditions. Since it is thermally beneficial to cant the engines, the recommendation is to allow 20-degree angles on roll and retro-axial engines, and no cant angle on the forward-axial engines.

4.3.6 Conclusions

As a result of the guidance and control study, a number of significant conclusions were reached concerning requirements in the areas of guidance and navigation, thrust vector control and the auxiliary propulsion system. Conclusions for each area are as follows:

Guidance and Navigation

1. The delta velocity budget furnished by MSFC/NASA is valid for the baseline mission.
2. The estimated impulse required for the main propulsion system is 25.4×10^6 lb-sec. That for the reaction jets is 430,000 lb-sec, including both translational and rotational usage.
3. The baseline thrust level of 10,000 lb for the main engine is acceptable for guidance and control functions.
4. The required inertial instruments (gyros and accelerometers) performance is well within the state of the art. Likewise, the star and horizon sensors required are also within the state of the art.
5. The required Laser is within the specification of the International Telephone and Telegraph that is under development.
6. For closing the 6000 n.mi. separation between the Tug and Satellite/Return-Payload, orbit transit time of 72 hours is needed to meet the delta velocity allocated by MSFC. In this connection, orbit transit time in multiples of 24 hours represents minima for delta velocity required. Maxima for required delta velocity occur at mid-point of each 24 hours interval. At the specified 6000 n.mi. separation in co-orbit condition, the sensitivity of delta velocity to separation distance is 1.6 fps per 100 n.mi.

7. At the NASA-specified three sigma navigation accuracy level, the navigation error causes an additional delta velocity of 70 to 150 ft/sec during rendezvous. This range applies nearly equally to 100 n.mi., 270 n.mi. and geosynchronous orbits depending on "a priori" spatial error distribution. These distributions are characterized by combinations of plus or minus velocity and positional errors.

8. The concept of using star and horizon sensor data for updating state vector is proposed for the Tug.

9. As a result of the error analysis performed for the powered phases, about 14 fps (3-σ) velocity error is accumulated for the 100 n.mi. x 19323 n.mi. orbital injection burn. The corresponding position error is only 9000 ft. Errors for all other burns are considerably smaller. Because of the 14 fps error, its cancellation is recommended once star and horizon sensor data are processed for a state vector update after the burn. If not corrected, the trajectory dispersion after 5-1/2 hours free fall to synchronous altitude would be unacceptable. Additionally, at least two other mid-course corrections are recommended. The last of the mid-course corrections should occur just prior to the circularization and 26.5 deg. plane change burn at 19323 n.mi. orbit apogee.

Thrust Vector Control

1. Rate stabilization back-up mode of the vehicle is recommended for event of loss in attitude control signal commands during main propulsion system thrusting.

2. The recommended TVC requirements for the Engine Actuation System are: ± 7 deg. gimbal deflection per axis, 8 deg/sec gimbal rate, and 12 deg/sec^2 gimbal acceleration.

Auxiliary Propulsion System

1. Both high and low thrust requirements have been met with a system which is capable of performing midcourse delta-V corrections and large attitude maneuvers rapidly, while retaining the precision control capability necessary for active and passive docking. The system uses approximately 1.2 pound of propellant per hour for prolonged attitude stabilization periods. Fail safe redundancy in a minimum-weight system is accomplished with 14 engines, each associated with a set of linked start valves and linked prevalves. Independent access to a common propellant feed line manifold is provided for each engine. A single engine failure results, in the worst case, in the inability to control translation in the lateral (sideways or vertical) direction. The capability to perform passive docking stabilization and to translate axially is retained in the selected APS configuration after a single engine failure.

2. The engines are arranged in two diametrically opposite clusters of five engines each, and two clusters of two engines each, to form a symmetrical configuration. Each pentad cluster contains one engine oriented in the radial direction. A total of four engines are available for axial thrusting. Engine couples generate control torques in all axes. Although it is recommended that all APS engines be located at approximately the same body station, no specific location is preferred.

3. APS total impulse requirements were determined for four modes of operation: attitude hold (stabilization), attitude maneuvers, propellant settling translations, and delta-V translation maneuvers. These requirements were determined for the 3000, 4160, and 8060 lb payload mission cases. The total requirements were determined to be approximately 427,700, 247,300, and 184,700 lb-sec respectively. The design requirement is therefore recommended at 430,000 lb-sec.

References

(4-1) SD70-547, "Guidance and Navigation Techniques for Long Duration Manned Space Systems," November 1970.

(4-2) SAMSO-TR-71-238, Vol. IVB, "Final Report, Orbit-to-Orbit Shuttle Feasibility Study." Vol. IV, System Design, Appendix B "Avionics Studies of the Orbit-to-Orbit Shuttle," NR Space Division, October, 1971.

5.0 DESIGN CRITERIA

The Design Criteria and Constraints have been developed for use in planning the development of the Space Tug. This is derived primarily from missions to be flown, mission ground and orbital operations, safety/reliability considerations, environment in which it will operate and basic Tug configuration. Constraints will be imposed on the Tug by the Space Shuttle vehicle in the areas of size, geometry, and weight. Technology or manufacturing (tooling) development may also constrain Tug design.

The Criteria and Constraints imposed on Tug design are detailed in Appendix B of this report. Data source for development of the Design Criteria and Constraints include the Statement of Work Study Plans' "Guidelines and Assumptions," revisions and additions thereof, telecons with NASA Study Manager, past experience of space programs and studies, and design analysis of this study.

The study guidelines provided the data for mission description, Tug payload delivery and recovery requirements, the basic Tug configuration, subsystem requirements, and Space Shuttle constraints, i.e., size, weight, and geometry and the environment to which the Tug will be exposed while in the Space Shuttle Orbiter cargo bay.

Reliability criteria were developed during the study based upon past Apollo and Saturn experience and extrapolated for the 1976 technology. Safety criteria was developed as a result of mission and design analysis and defined safety criteria for the individual Tug subsystems as well as criteria common to all subsystems.

In addition to the natural and induced environments identified by the Statement of Work Design Guidelines and Assumptions, the Tug requirement for inerting the Orbiter cargo bay with gaseous nitrogen while on the launch pad and during ascent to orbit has been identified.

A loads sign convention and control axis coordinates and sign convention has been established to provide a common reference point for future study as well as Tug/Orbiter on Tug/Payload orientation reference.

The Tug is to be designed as a fail-safe vehicle which has no failure mode which could cause the payload to be destroyed or a mode which could cause an unsafe situation for the Space Shuttle or its crew. To meet these requirements, subsystem and system redundancy criteria has been defined.

Operations of maintenance, repair, and refurbishment will be done on the ground. In order to enable these operations to be accomplished effectively, design guidelines have been established to facilitate test and inspection with due concern for the vital need to minimize Tug weight. These guidelines include the latest development in nondestructive test techniques.

Maintainability is a design characteristic which permits servicing, checkout, fault detection and isolation, repair, replacement and refurbishment of the Tug subsystems and components in an efficient and cost effective manner. Maintainability criteria for the Tug have been identified in general, installation, and subsystem categories.

Transportation and ground handling criteria take into consideration the operations involved in initial shipment of the Tug from the manufacturing site to eventual installation of the Tug in the Space Shuttle Orbiter cargo bay. Design criteria for the structure, avionics, propulsion and thermal subsystems were based on the Statement-of-Work "Design Guidelines and Assumptions." These data were modified and augmented by additional criteria resulting from the study design analysis activity.

Interface criteria includes physical and functional aspects of Tug/Shuttle, Tug/Payload, Tug/GSE, Tug/Engine and Tug/GSE via Shuttle interfaces.

Ground Support Equipment becomes an extension of the Tug systems during checkout, maintenance, servicing, handling and transportation of the Tug. GSE related design criteria is identified for implementation in the Tug design wherever possible without inducing and appreciable weight penalty to the Tug.

Although the facilities and manufacturing tooling are designed to accommodate the requirements of the Tug, they may impose constraints on Tug design due to tooling technology development or facility cost or availability factors. No constraints of this nature have been identified in this study, but the items "Facilities" and "Manufacturing" are included in the Criteria and Constraints appendix to preclude an inadvertent omission of these areas in follow-on studies.

6.0 INTERFACE CONTROL DOCUMENTS

As part of the overall system engineering process of ensuring a coherent Tug design to satisfy the specified mission objectives, five critical interfaces were identified, and their individual specifications presented in Appendix C.

The five interfaces identified are:

a. Tug/Shuttle

b. Tug/Main Engine

c. Tug/Payload

d. Tug/GSE for Servicing via Shuttle/GSE

e. Tug/GSE for direct Tug servicing

The intent of these preliminary Interface Control Documents (ICD's) is to define the full scope of interfaces which must be controlled throughout the development, manufacture, checkout and operational processes of the Tug subsystems. Since ICD's must be negotiated between affected contractors, and ICD's prepared during this time period can only be considered as recommended criteria for the next development phase. Each ICD was structured so as to permit easy growth into a document suitable for use as an instrument for interface control. All pertinent data, including physical, functional and procedural requirements, believed necessary to control the interface has been identified, and where data are not available blanks are provided so that data can be filled in at a later time. Since no data were provided by the NASA regarding the Shuttle and Mission payload configurations for the Tug study, interface requirements for these interfaces were necessarily general.

VOL II

APPENDIX A

FUNCTION AND OPERATIONS ANALYSIS

FUNCTION AND OPERATIONS ANALYSIS

1.0 INTRODUCTION

The primary objective of the space tug study is to determine the feasibility of a design point having a subsystem mass fraction of 0.895 and capable of ascending to a geosynchronous orbit from a 100 n.m. shuttle operations orbit with the following payload conditions:

a. Deliver and return a 3000-lb payload.

b. Deliver a 8060-lb payload and return empty.

c. Deliver no payload but return a 4160-lb payload.

To ensure that a coherent total system design is provided, a function and operations analysis was conducted and the results are presented herein.

The generation of system requirements for the space tug requires the examination of the development and operational cycle of the Space Transportation System (STS), particularly as it relates to the Tug. The STS as defined herein includes the booster, orbiter and Tug.

A key step in generating system requirements is a definition of the functions which must be performed by the Tug to accomplish the prescribed mission objectives. Likewise, a key step in assessing the relationship of the Tug to the STS development and operational cycle is an identification of top-level functions that would influence Tug design and operation. Thus, the function and operations analysis was directed toward the production of data on which to base system requirements, constraints, and other factors influencing the Tug design.

It is to be noted that the time required to carry out a particular function cannot be determined directly from a functional flow but is the result of a timeline analysis. What the functional flow does fix is the identification and the sequence of events (or alternative sequences), and it does determine the order in which operational actions occur. Thus, functional analysis provides a base for timeline generation which, in turn, will furnish information essential to determination of any time constraints.

2.0 BASELINE MISSION

2.1 Top Level Functions

In examining the top-level functions ascribed to the space transportation system, 14 major functional flow blocks (FFB) were identified in the cycle of production, use, and reuse of STS elements. A functional flow block diagram (FFBD) showing the relationship of these top-level blocks is presented in Figure A-1.

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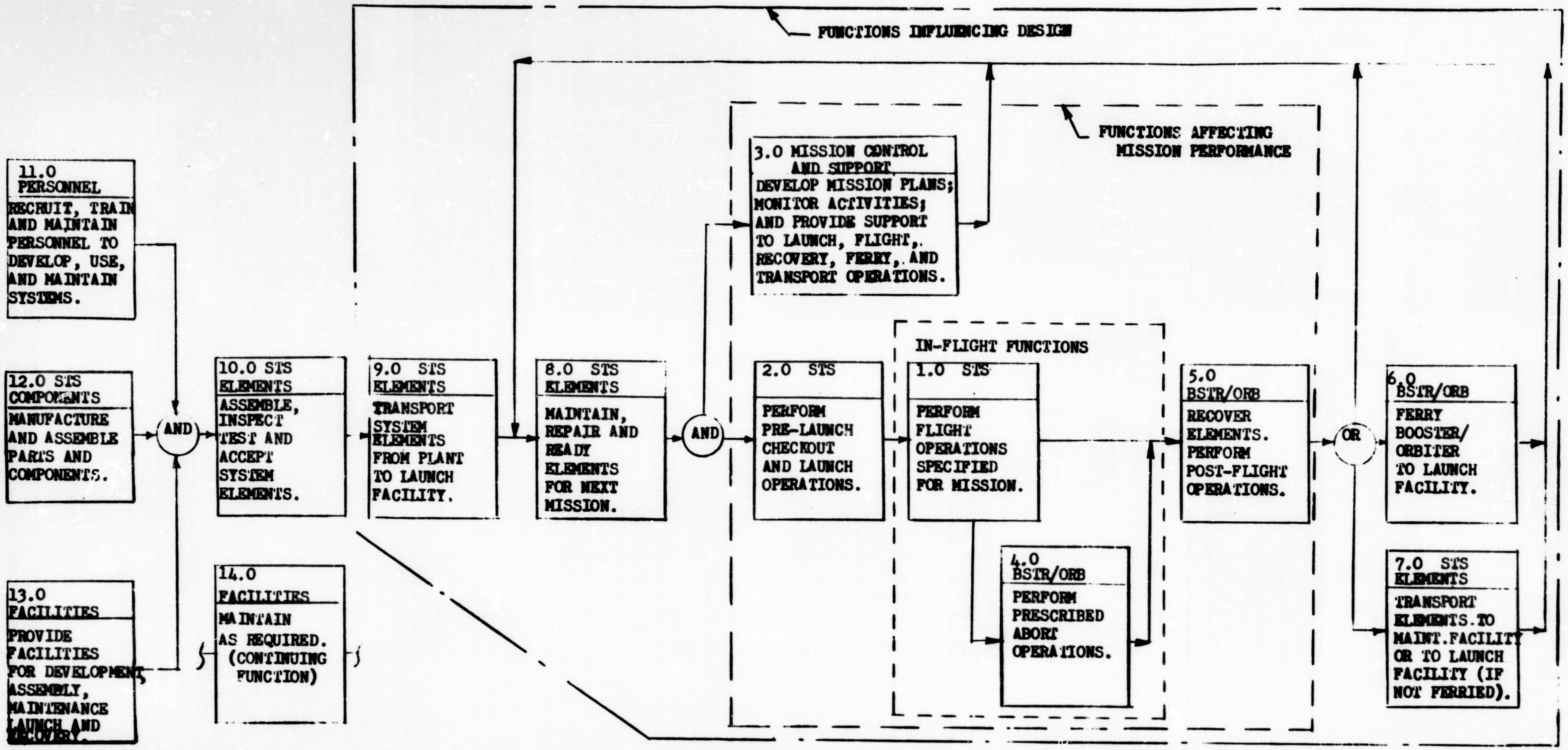

Figure A-1 TOP-LEVEL FUNCTIONS FOR THE SPACE TRANSPORT SYSTEM SHOWING AREAS AFFECTING DESIGN, PERFORMANCE AND IN-FLIGHT OPERATIONS.

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A separate block for the recovery function is shown since it occurs on the ground. Additionally, the assembly function is separated from the mission oriented function of the preflight checkout and launch. As a result of separating these functions, it is possible at the top level to isolate ground operations involved in inspection, maintenance, storage, handling, repair, and assembly of the Tug from the ground-based operations involved in mission control and support.

The initial phase of the development and operational cycle is, in the main, concerned with the three sets of functions relating to recruitment of personnel, manufacture of system components, and the development of facilities (FFB's 11.0, 12.0 and 13.0). These activities are followed by a block of functions involving the assembly, testing, and acceptance of system elements (FFB 10.0) and an intermittent but continuing function of maintaining the facilities developed for the program (FFB 14.0). These five sets of functions are not considered further in the analysis as they were deemed to have little, if any, influence on Tug design requirements.

Following acceptance, the functional flow moves to the transport of the system elements from the producing plant to the assembly/launch facility (FFB 9.0). The transport block is followed by a functional block covering the maintenance, repair, assembly, and readying of the systems for the next mission (FFB 8.0). It is at this point that reuse of system elements begins; therefore, there is a feedback loop to FFB 8.0 from the post-flight operations of recovery, ferrying, and transport as well as from the block designated Mission Control and Support.

Following readying for the mission, the functional flow divides into two branches. One goes to mission control and support and the other carries the STS through prelaunch and launch operations (FFB 2.0), flight operations (FFB 1.0), and recovery (FFB 5.0). Prelaunch, checkout, and launch operations comprise all those activities required to ready the STS for the mission up to actual liftoff.

Flight Operations (FFB 1.0) includes all those vehicle operations occurring between liftoff and return landing. It was these activities that were subjected to intensive functional analysis for each of the three design missions. A no-go functional block (FFB 4.0) is shown leading from flight operations to provide for abort operations. It is recognized that an abort might arise at any point during the flight phase of the mission. Since the Tug cannot return to the earth's surface by itself, the abort procedure, assuming the Tug has separated from the orbiter, incorporates a functional block requiring it to determine and carry out those operations which will return it to the orbiter, if that is possible. The Tug return to the orbiter is then referenced to the abort procedure of FFB 4.0. No further breakdown of the abort operations beyond FFB 4.0 is presented because it is beyond the scope of this analysis and also, it is essentially an orbiter function and not a Tug function.

At the top level, recovery operations are included in FFB 5.0, which follows directly after flight operations. Here, the term "recovery" is used only to designate that set of operations involving safing and securing the

booster, the orbiter, the Tug, and any payloads after landing. The term "retrieval" is reserved to designate those activities relating to repossession of a vehicle in space. For all design missions, normal flight operations of the orbiter end with landing at ETR and, thus, recovery of the orbiter, the Tug, and any payloads is accomplished at the launch facility and no transport or ferry operations (FFB's 6.0 and 7.0) are required. However, the booster could land at some point other than the launch site and, under contingency conditions, the orbiter could land at some point other than planned. In those situations, recovery of the booster and orbiter (without the Tug/payload) is accomplished using their own airbreathing engines for a direct return to the maintenance/assembly/launch facility. The Tub/payload, detached from the orbiter is then transported by other means to the maintenance facility. Since FFB 6.0 involves only the booster/orbiter, no further breakdown of this functional block will be made.

The branch paralleling flight operations and recovery is designated Mission Control and Support (FFB 3.0). Even though maximum autonomy is a design goal for Tug, an overall control and support function will exist. It will include activities such as development of mission plans and schedules, preparation of command data for each mission, establishment of criteria and decision logic for handling mission contingencies, monitoring and recording of actual mission events, performance of launch and recovery operations, and aid to any transport or ferrying activities involving return of elements to the assembly/launch site designated for their next use. This function, as with FFB's 4.0 and 6.0, does not involve specific Tug operations and is therefore discussed only in the top level diagram.

This means of accomplishing 9 of the 14 top level functions will have a direct influence on the design of the Tug. For example, handling and tie-down during transport either from the production plant or from the recovery point will affect location and design of attachment points of both the Tug and the orbiter. The remaining five functions are mission oriented, i.e., the manner in which they are accomplished is directly related to the objective to be achieved in a particular mission. Thus, these functions must be taken into account in evaluating the performance of each Tug design. In turn, two sets of functions involve in-flight activities from which the majority of the system requirements derive. The three major regions affecting the design, mission performance and in-flight operations are shown on the face of Figure A-1 as areas enclosed in dashed lines.

2.2 First Level Functions

First level functions comprising the block designated FFB 1.0 of the top level functional flow diagram vary according to the particular mission of the Tug. Three first-level function diagrams were generated, one each for the baseline mission, the delivery mission and the retrieval mission. The majority of the first-level functions are identical for the three missions. The difference occurs only in functionsl blocks involving the specific disposition of the Tug payload (i.e., deliver payload with/without the retrieval of another payload, etc.).

In the subsequent section, 16 functional flow block diagrams are used as aids in describing the first- and second-level functions required for the three Tug missions. To aid in locating the diagram that applies to a particular major function and mission, Table A-1 provides an index of the diagrams by mission and function category.

Table A-1. Index of Functional Flow Block Diagrams

Function Level and Objective	Figure Number of Diagram Pertaining to Mission		
	Baseline (Del & Retr)	Delivery Mission	Retrieval Mission
First-level functions			
Ground-based mode	3.1-1	3.1-15	3.1-12
Second-level Functions			
Tug Deployment	3.1-2	3.1-2	3.1-2
Perform Coast Operations	3.1-3	3.1-3	3.1-3
Perform Ascent Ellipse Injection	3.1-4	3.1-4	3.1-4
Inject Into Geosynchronous Orbit	3.1-5	3.1-5	3.1-5
Deployment of Payload	3.1-6	3.1-6	N/A
Injection Into Retrieval Phasing Orbit	3.1-13	N/A	3.1-13
Retrieval of Payload	3.1-14	N/A	3.1-14
Injection Into Return Transfer (DEI) Orbit	3.1-7	3.1-7	3.1-7
Injection Into Return Phasing Orbit & MCC	3.1-8	3.1-8	3.1-8
Injection Into Operations Orbit	3.1-9	3.1-9	3.1-9
Perform Orbiter/Tug Rendezvous and Docking	3.1-10	3.1-10	3.1-10
Perform Post-Docking Orbit Coast Operations	3.1-11	3.1-11	3.1-11

It should be noted that, although a particular function may be common to two or more missions, it does not follow that the same activities are always carried out to perform the function, and more particularly, that the same timing or the same quantities pertain. For example, the direction of and the delta-V required for the outbound transfer orbit of the delivery mission may be different from those for the corresponding orbit of the retrieval mission, yet the same second-level functional flow applies.

The 18 first-level functions needed to accomplish the flight operations in the baseline mission are illustrated in Figure A-2. The flight operational sequence is assumed to start at liftoff at ETR. The STS, composed of a booster, an orbiter, and the Tug, performs those operations associated with the boost phase of the flight (FFB 1.1). At the designated separation point in the

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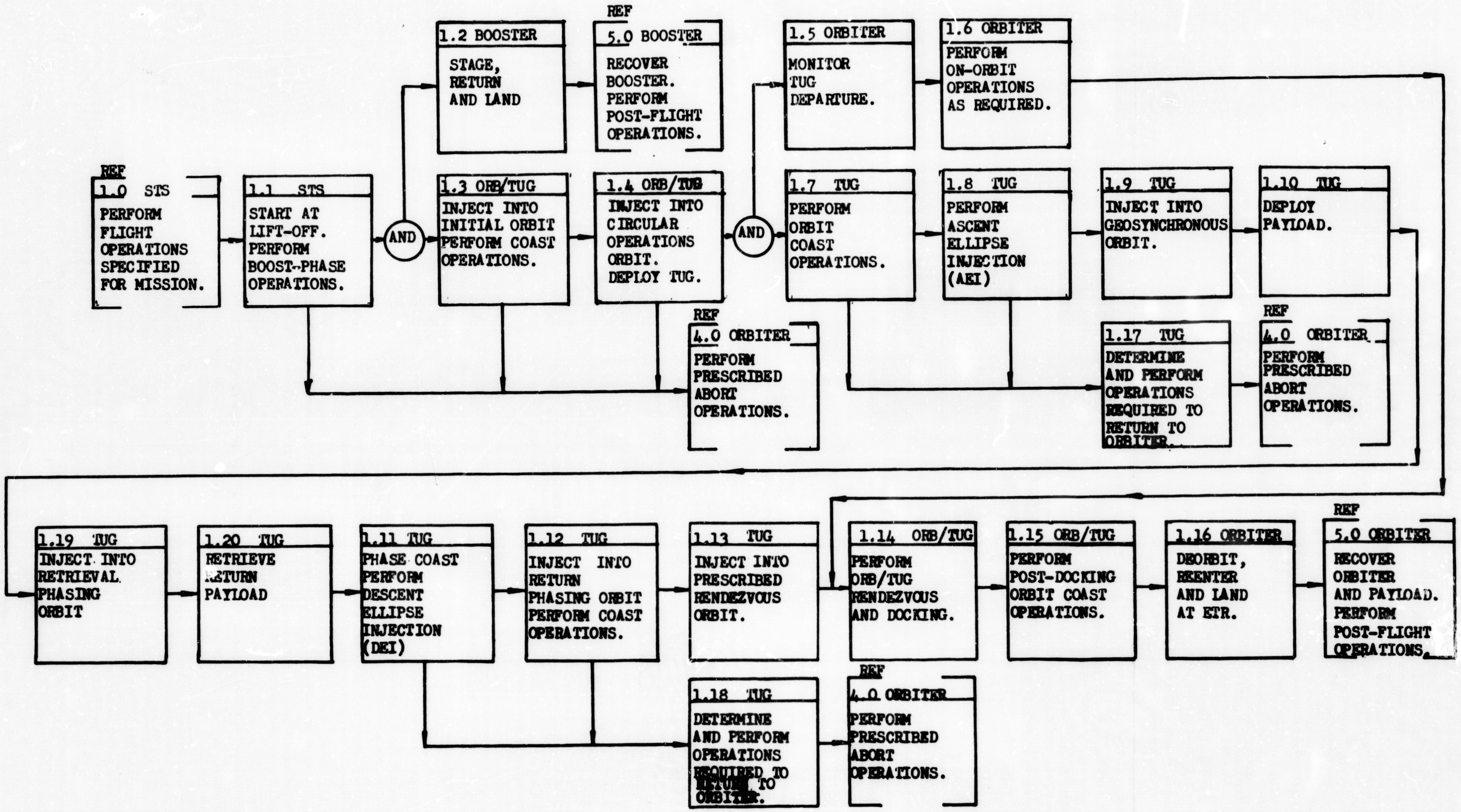

Figure A-2. FIRST-LEVEL FUNCTIONS FOR BASELINE MISSION

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trajectory, the booster stage then returns and lands at the planned recovery point (FFB 1.2). Booster landing may also be planned to occur at a down-range landing site, followed by a ferry flight back to the launch site. The orbiter with its payload (Tug plus mission payload) injects into an initial 50 n.m. by 100 n.m. orbit and performs the necessary stabilization and guidance operations (FFB 1.3). This is followed by the orbiter's insertion into the 100 n.m. circular operations orbit and subsequent deployment of the Tug (FFB 1.4). Until start of the deployment sequence, the Tug is in a quiescent state, and its subsystems are either shut down or operating at a standby level. Thus, the prinicpal design requirement imposed on the Tug is that the structural strength be sufficient to withstand the stress loads associated with launch and boost to orbit. No-go's of FFB 1.1, 1.3 and 1.4 are referenced to FFB 4.0, which provides that the orbiter will follow a preplanned procedure in the event of an abort.

Following separation from the orbiter and final check to verify mission readiness, the Tug injects into the outbound transfer orbit (FFB 1.8). The orbiter monitors the Tug departure from the operations orbit (FFB 1.5) and then proceeds to coast, performing drag make-up and other maneuvers as needed to remain in the operations orbit while awaiting the return of the Tug (FFB 1.6). Gross failure in achieving the outbound orbit or occurrence of some other contingency situation may necessitate abandonment of the mission. Thus, the no-go from FFB 1.7 and FFB 1.8 leads to a function characterized by determining what operations must be carried out to return to the orbiter, computing the associated parameters, and then attempting to perform those operations (FFB 1.17). Successful accomplishment of the return leads to top-level FFB 4.0 denoting that the orbiter would complete the abort sequence.

At apogee of the transfer orbit, the Tug injects into synchronous orbit (FFB 1.9) and proceeds to locate and maneuver to the orbital position for deployment of the mission payload (FFB 1.10). Since the relative positions of the Tug and the orbiter as well as their remaining impulse capabilities are known in advance, contingency operations can be preplanned and no separate first-level function of determining these operations is required. After deployment of the mission payload, the Tug is injected into the retrieval phasing orbit and then follows with the retrieval of the return payload. Following the acquisition of the return payload, the Tug/PL is then subject to an Descent Ellipse injection (FFB 1.11) for the return to the 270 n.mi. phasing orbit (FFB 1.12). Upon completion of performing all necessary coast operations while in the phasing orbit, the Tug/PL is injected into the operations orbit for rendezvous with the orbiter (FFB 1.13). No-go during the descent ellipse injection or phasing orbit insertion leads to FFB 1.18 and the referenced abort operations of FFB 4.0.

The Tug then cooperates with the orbiter in performing the rendezvous and docking (FFB 1.14) and carrying out any necessary post-docking operations such as venting (FFB 1.15). The Tug assumes a quiescent state for its return while the orbiter deorbits, reenters, and lands at ETR (FFB 1.16). This final flight phase operation is referenced to top-level FFB 5.0 for recovery of the orbiter and the Tug.

2.3 Second Level Functions

The second-level functions comprising FFB 1.4 in the first-level functional flow diagram are displayed in Figure A-3. While the orbiter is in the initial orbit, it aligns its IMU and updates the state vector (FFB 1.4.1). It then determines the burn parameters and orients itself for insertion into the operations orbit (FFB 1.4.2). Insertion is performed by a burn of the OMS engines (FFB 1.4.3). The orbiter then coasts in orbit (FFB 1.4.4) and performs any necessary passive temperature control procedure (FFB 1.4.5) until it reaches the planned point for initiating the Tug deployment sequence. It then orients for deployment (FFB 1.4.6), and opens the cargo bay (the cargo bay may be opened any time after operations orbit insertion). The Tug subsystems, including the propulsion subsystems pre-start functions, are activated and checked out (FFB 1.4.7). Following the satisfactory check of the Tug subsystems, the Tug is deployed in position for separation (FFB 1.4.8). Safety is a consideration here. For example, activation of the Tug stabilization subsystem could possibly generate a torque or load which would damage the deployment/retrieval mechanism or even one of the vehicles. The activation and checkout function may overlap the preceding and following functions as its accomplishment will extend over an indeterminate period, depending upon the specific characteristics of the system components and the techniques used for equipment warmup and test. The initialization procedure for the Tug GN&C system will rely on supplied data in that the system will receive the necessary command and state vector data from the orbiter. This gives rise to the problems of compatibility of the Tug GN&C system with that in the orbiter and the means for transfer of accurate data in a timely fashion.

Following satisfactory checkout, the orbiter uncouples the Tug and separates by performing an ACPS burn (FFB 1.4.9). After the orbiter has moved a safe distance (300 meters) away, the Tug checks both its main engines and its attitude control propulsion system (ACPS) and carries out a final check as to its mission readiness (FFB 1.4.10). The orbiter also performs a post-separation check of its own subsystems and monitors for verification of satisfactory Tug final checkout (FFB 1.4.11). If mission readiness is not satisfactorily verified, the orbiter can retrieve it (FFB 1.4.12) and then proceed with a preplanned abort procedure (FFB 4.0). This contingency imposes a requirement that the Tug be capable of redocking with the orbiter with the mission payload still attached. If the Tug confirms readiness for departure on its mission, the orbiter retracts and stows the deployment mechanism (FFB 1.4.13) and continues to support and monitor the Tug during the performance of Tug coast operations and its preparation for the ascent ellipse injection phase.

Figure A-4 presents the flow of second-level functions associated with the orbit coast operations. The first function which the Tug must accomplish to prepare for the injection (AEI) burn is alignment of its IMU and determination of a current state vector (FFB 1.7.1). It then compares its orbital parameters with data received from the orbiter (FFB 1.7.2). If the two sets of parameters do not agree (i.e., within allowable tolerances), the Tug checks the inputs to and operations of its computer (FFB 1.7.3) and then loops back to perform the operations in FFB 1.7.1 again. If satisfactory agreement is

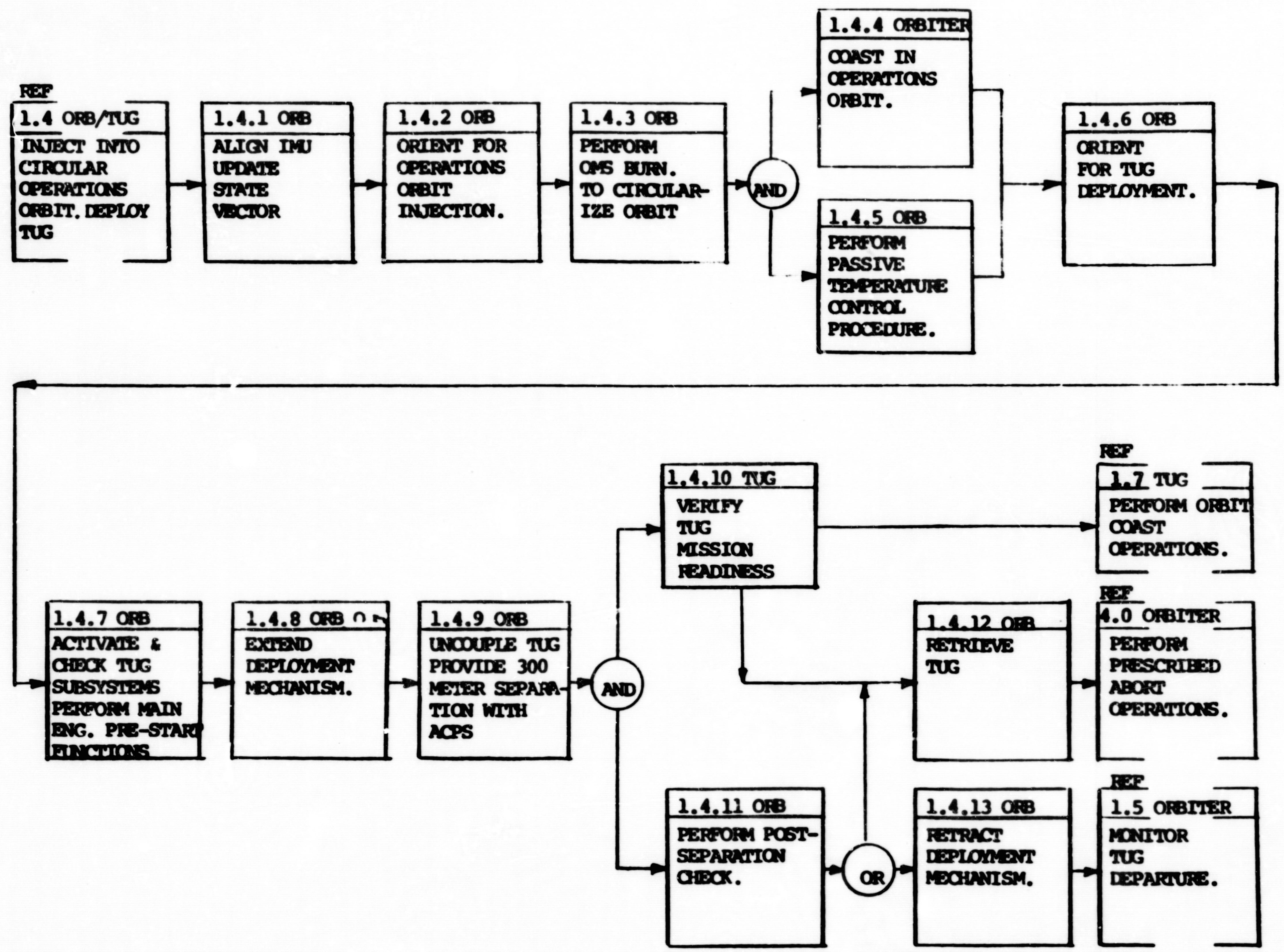

Figure A-3. Second-Level Functions - TUG Deployment

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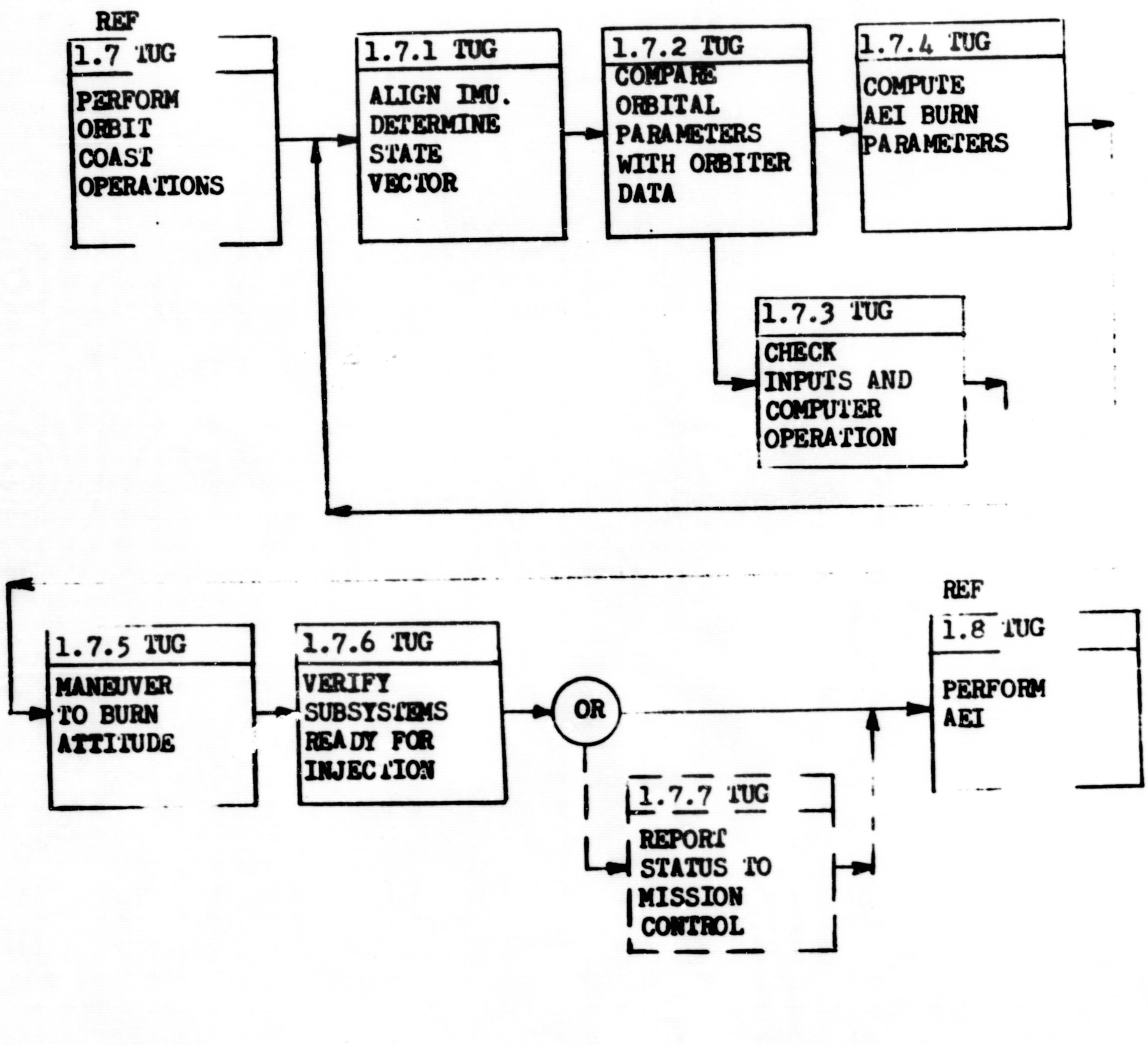

Figure A-4. SECOND-LEVEL FUNCTIONS - PERFORM COAST OPERATIONS

reached, the Tug proceeds to compute the ascent ellipse injection burn parameters (FFB 1.7.4), maneuvers to burn attitude (FFB 1.7.5), and verifies that its subsystems are ready for the injection (AEI) burn (FFB 1.7.6). An optional block (FFB 1.7.7) is inserted into the functional sequence to permit reporting status to mission control (or recording status for subsequent downlinking when within line-of-sight of a ground station). It was assumed that, even though the Tug operates autonomously, mission control might elect to receive status reports to keep informed on the progress of the mission and to provide a data base for executive override in the event a contingency situation should arise. The Tug then coasts in orbit until time to perform the AEI into the outbound transfer orbit (refer to FFB 1.8).

Nine second-level functions are identified with the ascent ellipse injection into the outbound transfer orbit and midcourse correction. They are diagrammed in Figure A-5. Using the sensing data course and corrections effected during the coast operation (FFB 1.7), the main engine is fired for the ascent ellipse injection (FFB 1.8.1). The injection sequence is completed after an assessment of the post-burn state vector is made (FFB 1.8.2).

At a predetermined point, the Tug makes a diagnostic check of its subsystems and decides whether any midcourse correction is required (FFB 1.8.3). Status could be reported to mission control following this check (FFB 1.8.4) or delayed (FFB 1.8.6) until after the correction burn parameters are computed (FFB 1.8.5). The course correction would be accomplished if required with an attitude control propulsion system (ACPS) engine firing (FFB 1.8.7), the post-burn state vector assessed (FFB 1.8.8), and, optionally, status reported to mission control (FFB 1.8.9).

The second level functions associated with injection into geosynchronous orbit are similar to those described for the injection into an outbound transfer orbit. Figure A-6 illustrates the second-level functional flow for insertion into geosynchronous orbit. No course correction activities are shown in this diagram as they are considered part of the payload deployment sequence which follows.

There are nine required functions plus three optional and two contingency functions involved in deployment of the mission payload (Figure A-7). Following an update of the state vector (FFB 1.10.1), the Tug GN&C system determines the degree to which the attained geosynchronous orbit varies from that planned for the payload (FFB 1.10.2). If the variance exceeds that allowed for the payload mission, the Tug computes the burn parameters to correct its orbit (FFB 1.10.3), reports status (optional FFB 1.10.4), and performs an ACPS burn (FFB 1.10.5). The functional flow then loops back to FFB 1.10.2 where the orbit variance is recomputed. When it falls within the prescribed limits, the Tug reports status (optional FFB 1.10.6) and maneuvers to the deployment attitude (FFB 1.10.7).

After physically detaching the payload (FFB 1.10.8), the Tug maneuvers to a safe distance (FFB 1.10.9) and stationkeeps with the payload while awaiting confirmation that the payload is to remain on mission orbit (FFB 1.10.10). This confirmation does not necessarily require the Tug to monitor the mission

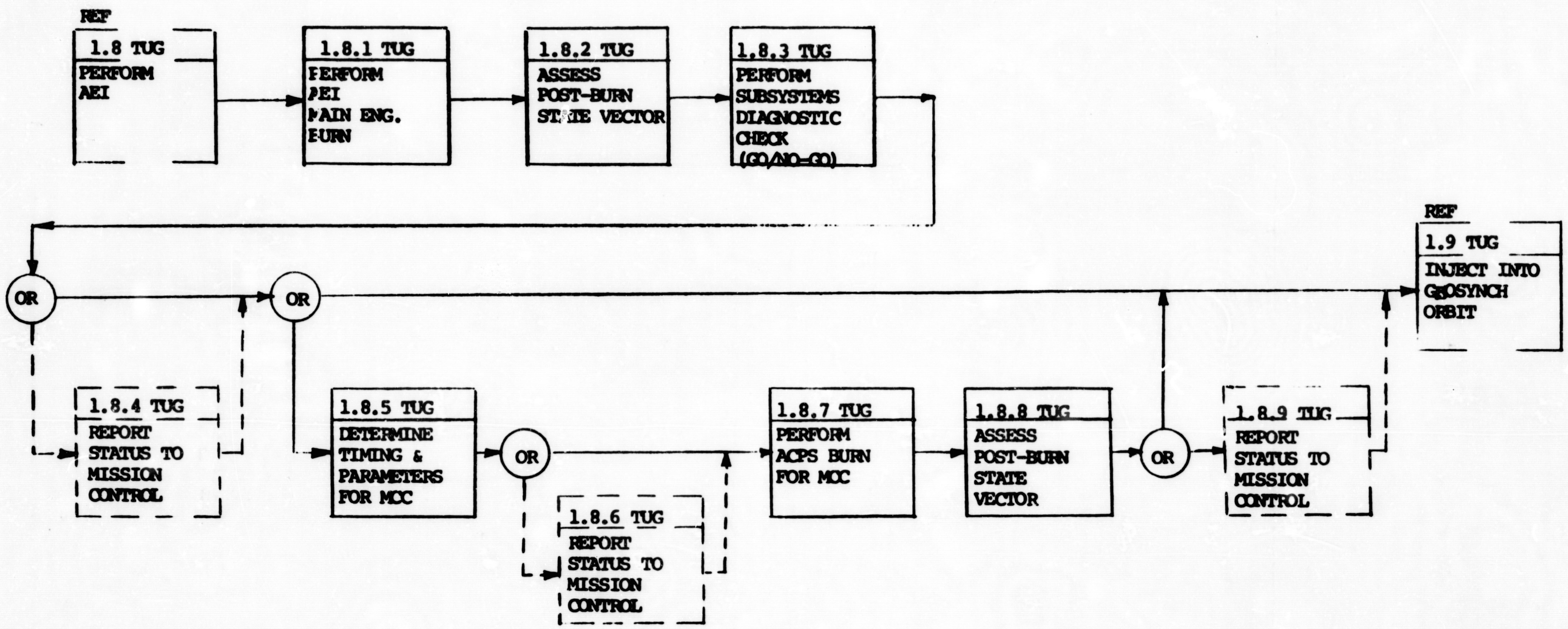

Figure A-5.. Second Level Functions - Perform Ascent Ellipse Injection (AEI)

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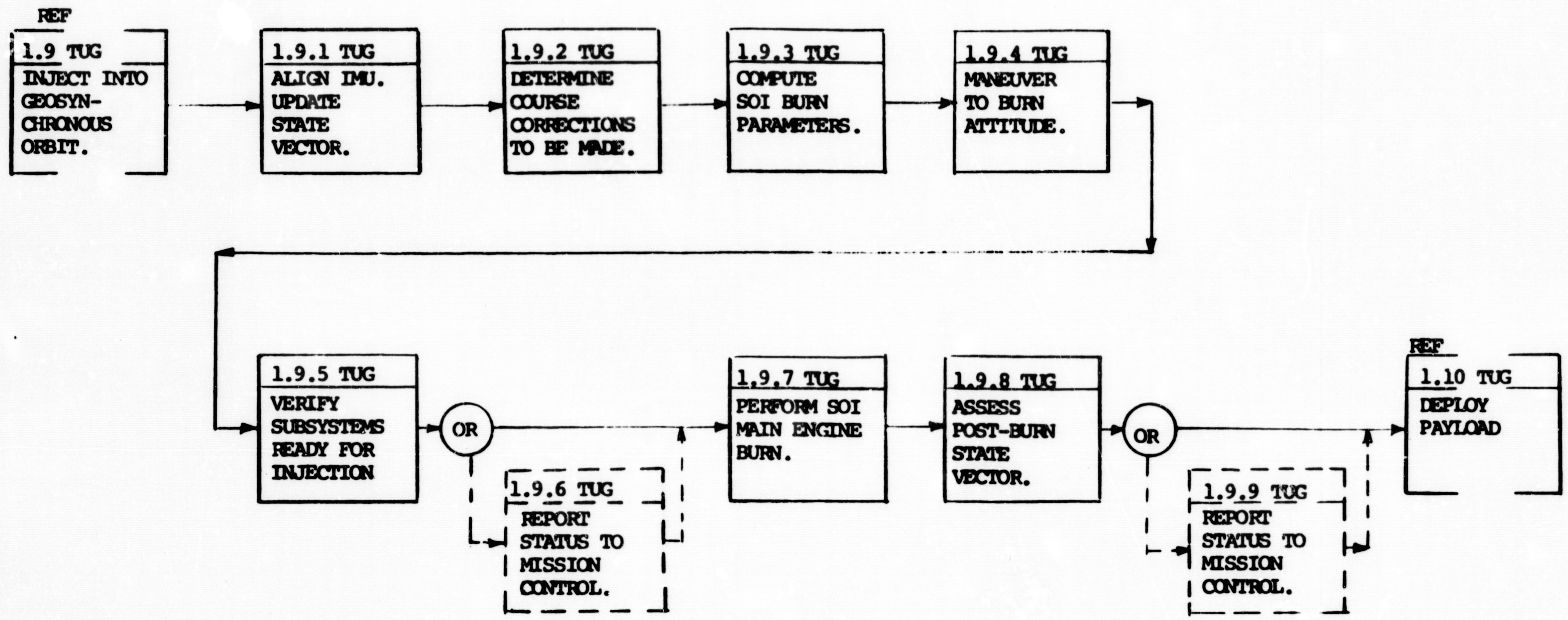

Figure A-6. Second-Level Functions - Injection Into Geosynchronous Orbit

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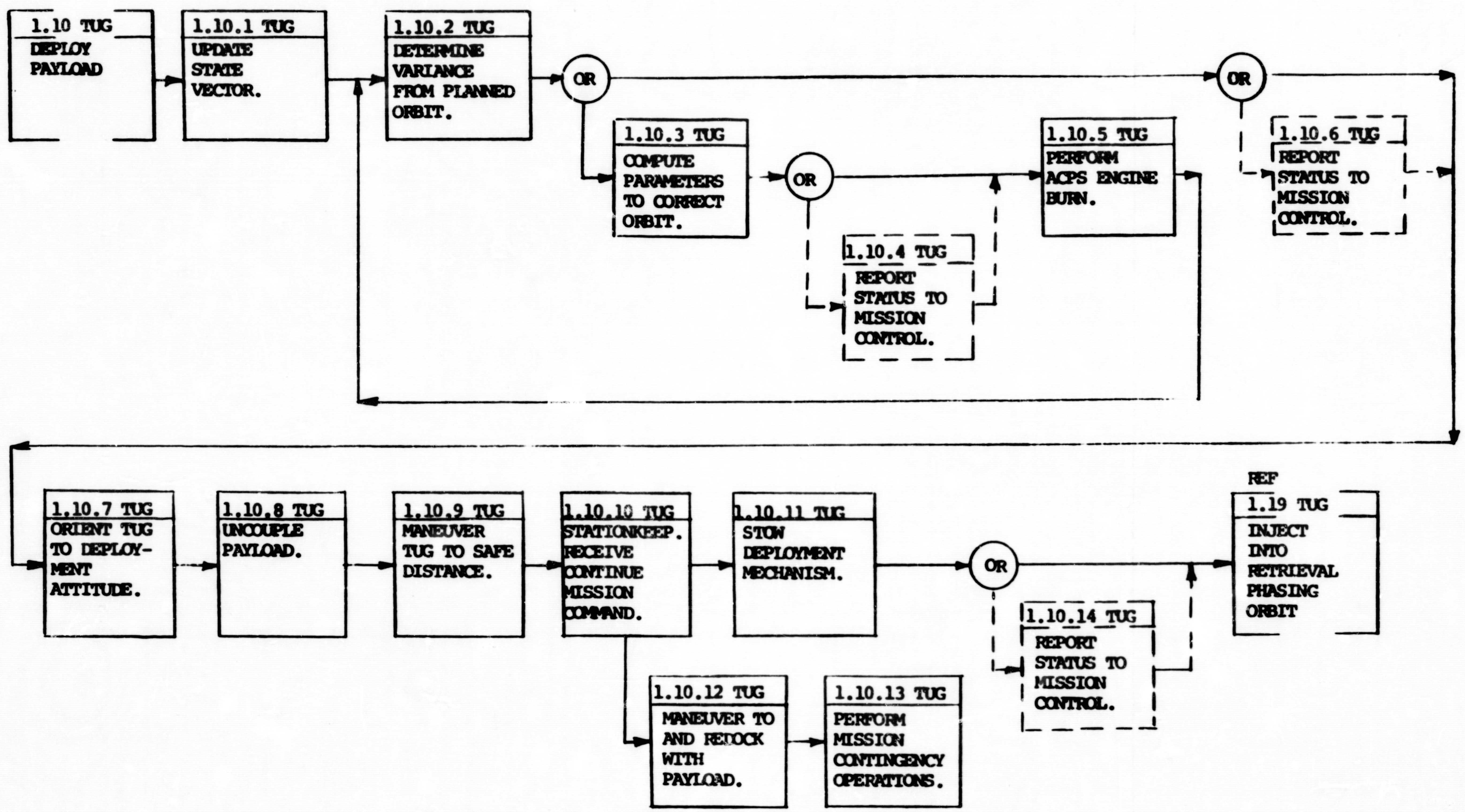

Figure A-7. Second-Level Functions - Deployment of Payload

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payload operations or receive signals from the payload vehicle. The confirming command might come from mission control or the Tug might be programmed to continue the planned mission, if it did not receive a command to initiate contingency operations within some definite period after payload release. If a contingency does arise, the Tug maneuvers to and retrieves the payload (FFB 1.10.12) and then performs any preplanned operations such as return of itself and payload to some lower orbit for subsequent retrieval by another STS mission (FFB 1.10.13). If payload operation is confirmed, the Tug stows and secures the deployment mechanism as necessary (FFB 1.10.11), reports status (optional FFB 1.10.14), and proceeds to inject into the retrieval phasing orbit (Reference FFD 1.19) to pick up another payload for the return trip to the Shuttle orbiter.

Figure A-8 displays the second-level functions involved in the Tug inserting itself into a retrieval phasing orbit preparatory to rendezvous with the payload to be retrieved. The functional sequence is similar to that for injection into any orbit. After aligning the IMU and updating its state vector (FFB 1.19.1), the Tug determines the course corrections to be made (FFB 1.19.2) during the orbit insertion. The GN&C subsystem then computes the burn parameters (FFB 1.19.3) and the vehicle is placed in the proper burn attitude (FFB 1.19.4). After verifying that its subsystems are ready for orbit injection (FFB 1.19.5), the Tug ACPS engines are fired (FFB 1.19.7). An assessment of the state vector is made (FFB 1.19.8) and the Tug main engine is fired to rendezvous with the return payload (FFB 1.19.9). The Tug then orients itself so that the sensors are directed toward the sector containing the payload to be retrieved (FFB 1.19.10). Optional blocks are shown which would permit report of operating status to mission control before and after the phasing orbit burn.

The functional sequence involving the rendezvous and retrieval of the return payload is shown in Figure A-9. It is assumed that once the Tug is in the retrieval phasing orbit, it will have sufficient data stored in the computer memory to determine which sector of space to concentrate the search. The first step in the functional sequence is for the Tug sensor to acquire and lock-on to the payload satellite (FFB 1.20.1), which will be equipped with passive tracking aids to assist the Tug in this operation. The Tug then computes the relative range between it and the payload satellite and the rate at which the range is changing (FFB 1.20.2). Through returns from the navigation aids, the Tug also determines the orientation of the satellite and, thus, the location of the docking mechanism (FFB 1.20.3). From these two sets of data (plus knowledge of its own state vector), the Tug determines what maneuvers are required for it to intercept the satellite's course in a manner that will permit proper alignment and docking (FFB 1.20.4). The Tug computes the initial terminal phase (TPI) burn parameters (FFB 1.20.5), verifies that its subsystems are ready for terminal phase burn (FFB 1.20.6), assumes the proper attitude, and performs an ACPS engine firing to effect the computed TPI maneuver.

Through midcourse of the terminal phase, the GN&C determines the amount by which the Tug would miss the planned intercept (FFB 1.20.9), computes the

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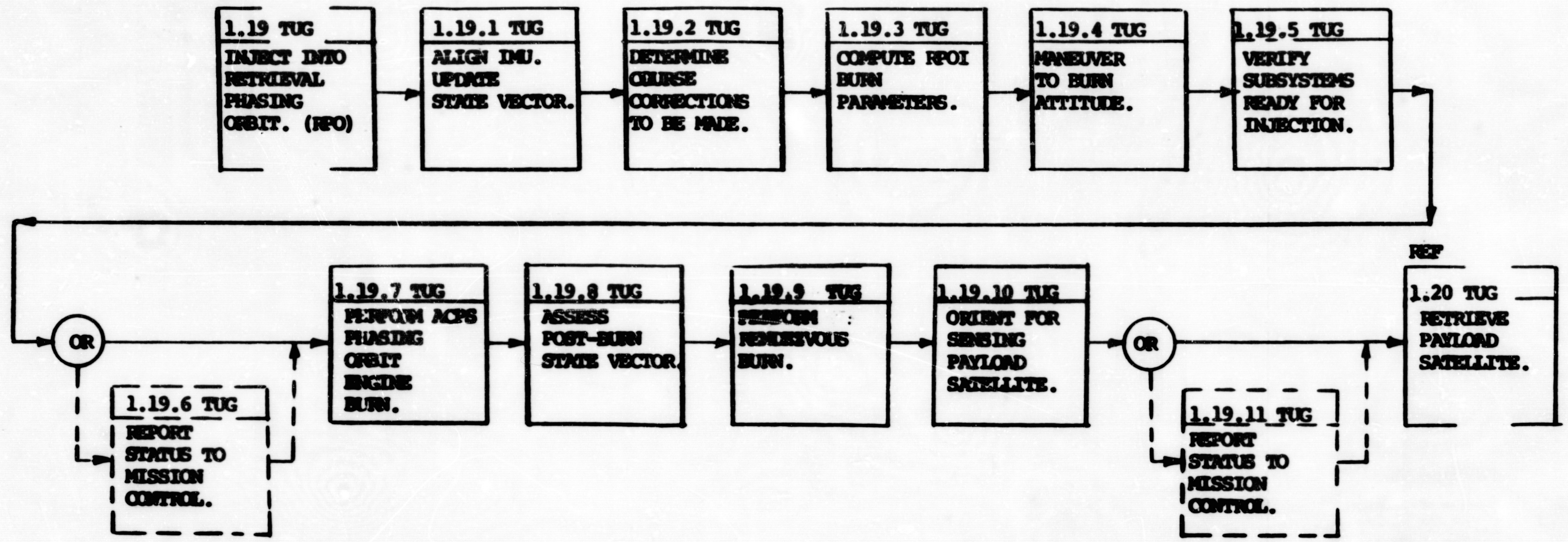

FIGURE A-8. Second-Level Functions - Injection Into Retrieval Phasing Orbit

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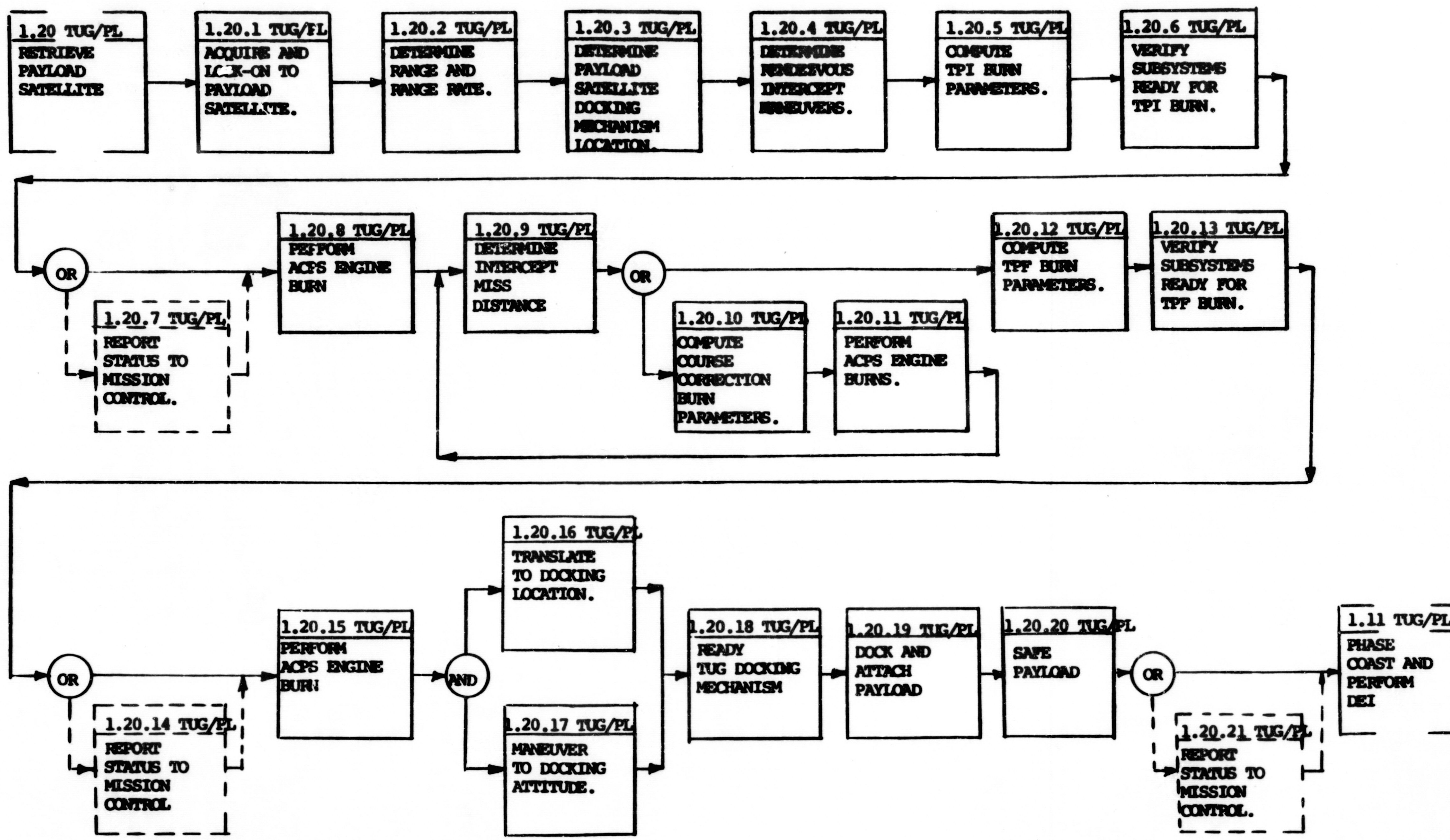

Figure A-9. Second-Level Functions - Retrieval of Payload

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burn parameters for any corrections (FFB 1.20.10), and recomputes the intercept miss distance. A final burn (TPF) of the ACPS engines is used to reach the point where closure begins, is typical for all rendezvous and docking functions and is shown in FFB's 1.20.16 through 1.20.19. Upon completion of docking and attaching the payload to the Tug, the payload is safed to ensure a successful return to the orbiter. (FFB 1.20.20).

Figures A-10 and A-11 show the second-level functions associated with injection into the return transfer (DEI) and return phasing orbits, respectively. The functions differ little from those described for the outbound orbits; only the sequence has changed, primarily because of the inclusion of the phasing orbit on the return flight. Likewise, the functions identified with the Tug injection into the operations orbit used for rendezvous with the orbiter (Figure A-12) are similar to those required for injection into any other orbit.

Normal orbiter/Tug rendezvous is accomplished by use of a modified co-elliptic technique in which the orbiter, as the active element, senses the position of the Tug relative to itself and flys an intercept course. The Tug cooperates with the orbiter by maintaining attitude, by providing a means for the orbiter sensors to determine its position and attitude, and, if necessary, it can become the active vehicle on demand. The second-level functions involved in rendezvous and docking are displayed in Figure A-13. After the orbiter sensors have acquired and locked on to the Tug (FFB 1.14.1), a communication link between the two vehicles is established (FFB 1.14.2), and control of Tug operations is assumed by the orbiter crew (FFB 1.14.3). While the Tug is maneuvering via the TPI and TPF burns into position and then maintains its attitude hold (FFB 1.14.4, 1.14.5, 1.14.6), the orbiter also performs the terminal phase maneuvers to bring it to the prescribed position for starting closure (FFB's 1.14.7, 8, and 9). The orbiter then performs both translation and attitude maneuvers in closing on the Tug (FFB's 1.14.10 and 1.14.11). The orbiter deploys the retrieval mechanism (FFB 1.14.12) and docks with and attaches the Tug (FFB 1.14.13). The combined vehicles then perform the necessary post-docking operations required before de-orbit (refer to FFB 1.15).

Five functions are identified (Figure A-14) for the post-docking orbit coast operations. Following the successful docking of the Tug to the orbiter (FFB 1.14.13), the Tug is stowed away in the cargo bay (FFB 1.15.1) and all disconnects remated (FFB 1.15.2). To ensure a safe return of the Shuttle/Tug to earth, all propellant tanks are emptied and inerted (FFB 1.15.3) while the Tug is in the Shuttle cargo bay. The safing sequence involves (1) dump propellant, (2) pressurize propellant tanks with helium gas, (3) vent helium/propellant gas mixture and, (4) repressurize tanks with helium, and hold at the pressure for deorbit and landing. After the safing operations are completed, all subsystems are deactivated for the deorbit operations (FFB 1.16).

3.0 DELIVERY MISSION

The first level functions associated with the delivery mission are shown in Figure A-15. It is noted that this flow is identical to the first level

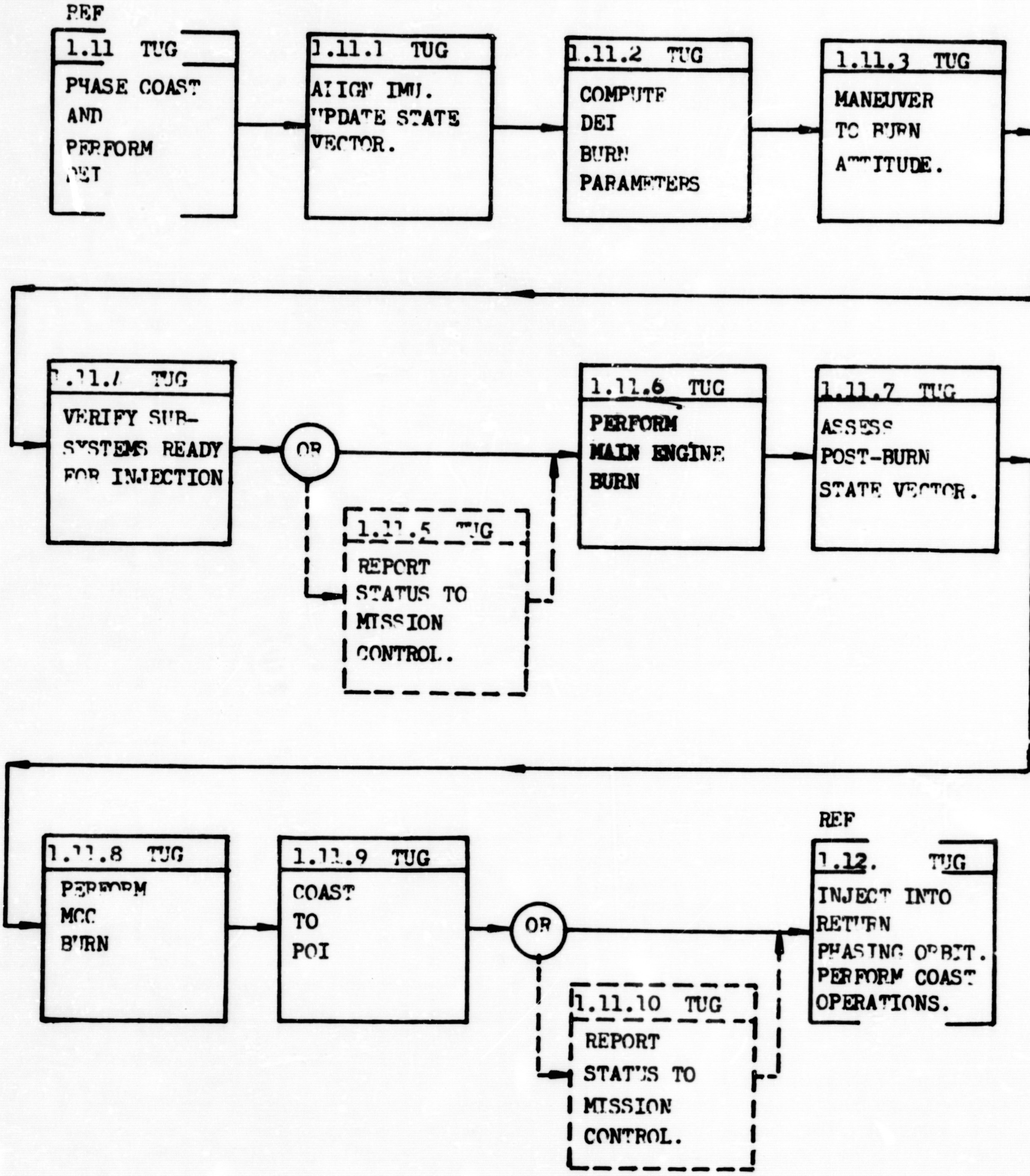

Figure A-10. Second-Level Functions - Injection Into Return Transfer Orbit

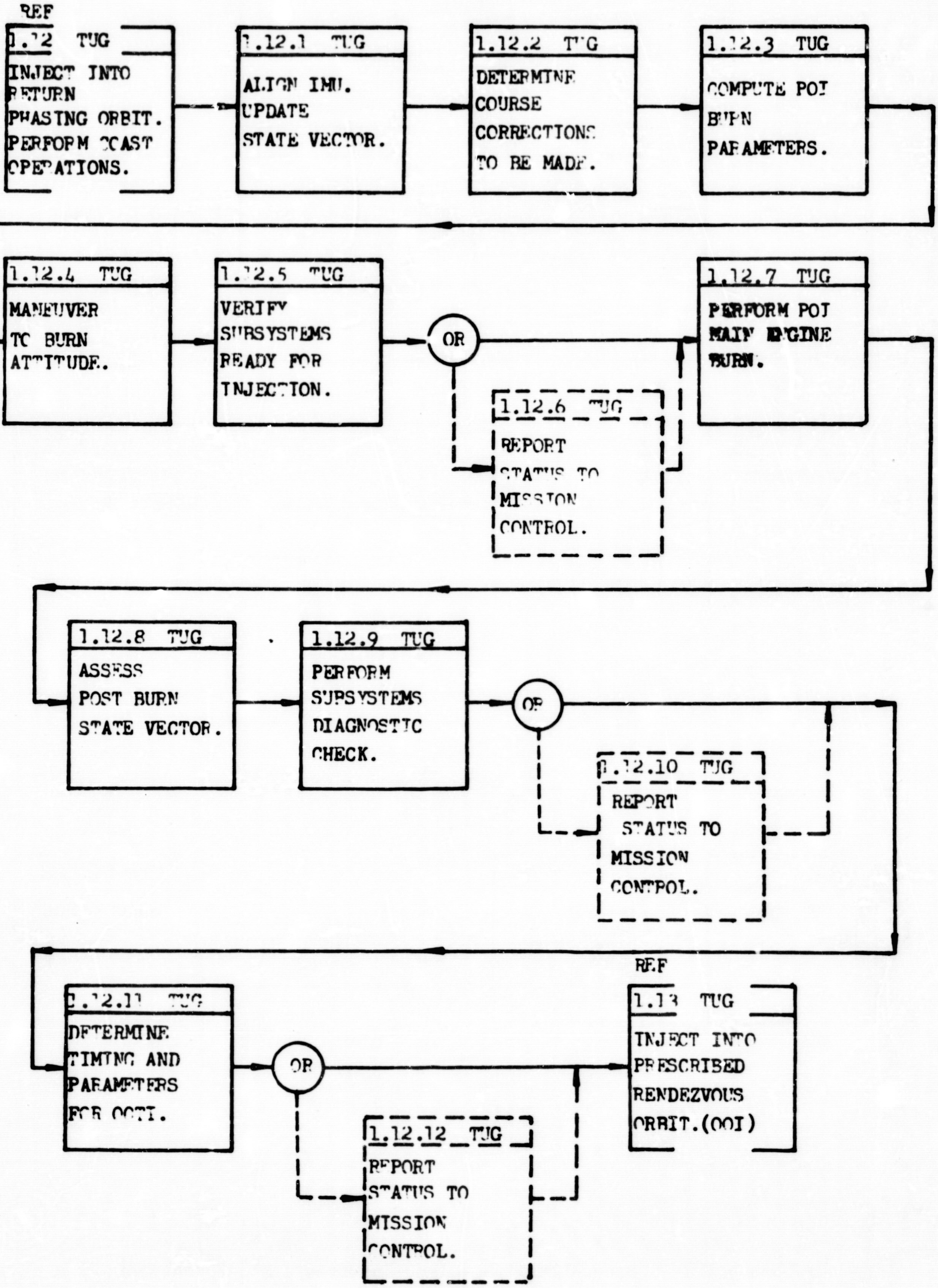

Figure A-11. Second-Level Functions - Injection Into Return Phasing Orbit and Midcourse Correction

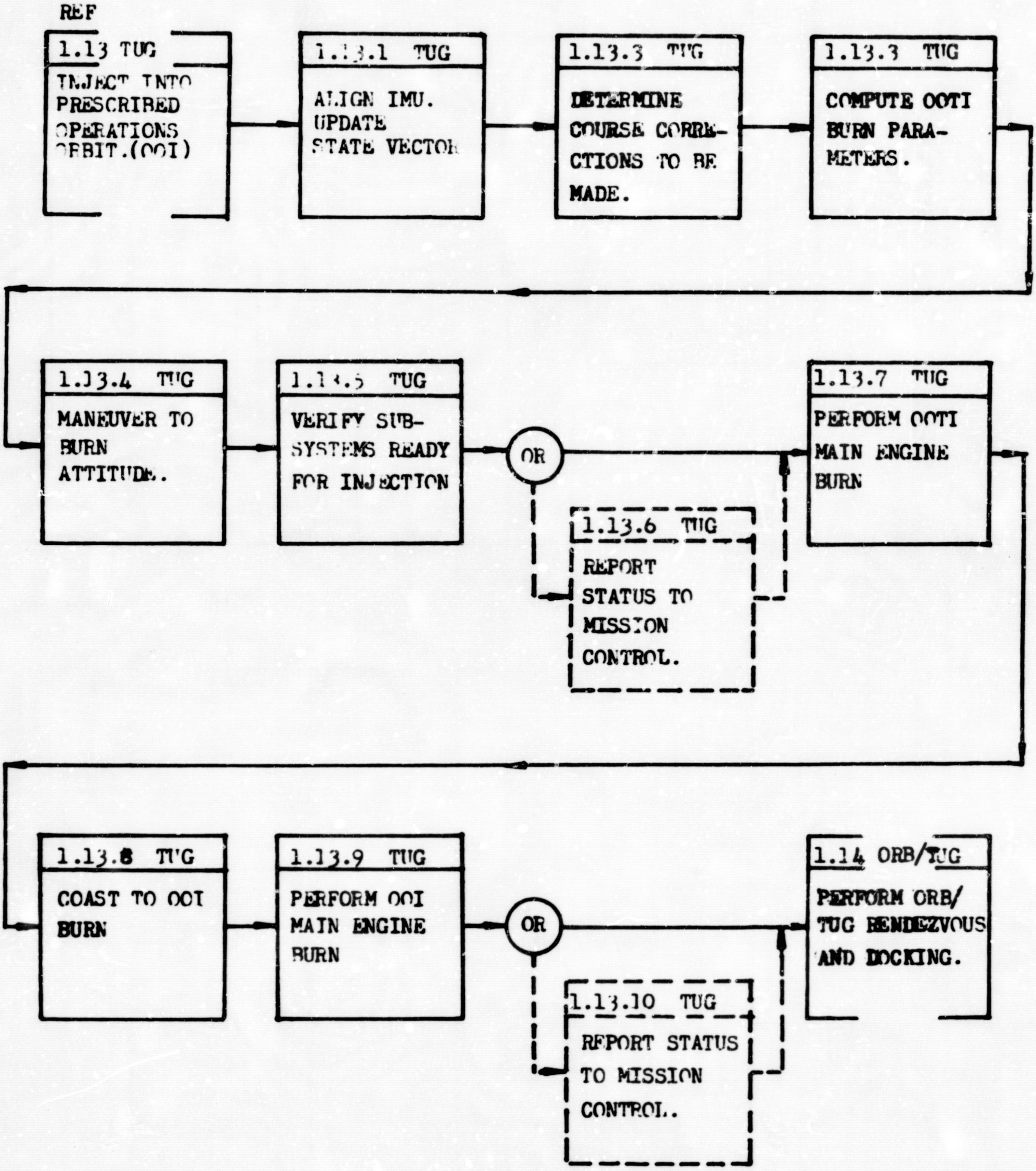

Figure A-12. Second-Level Functions-Injection Into Operations Orbit

(All Missions)

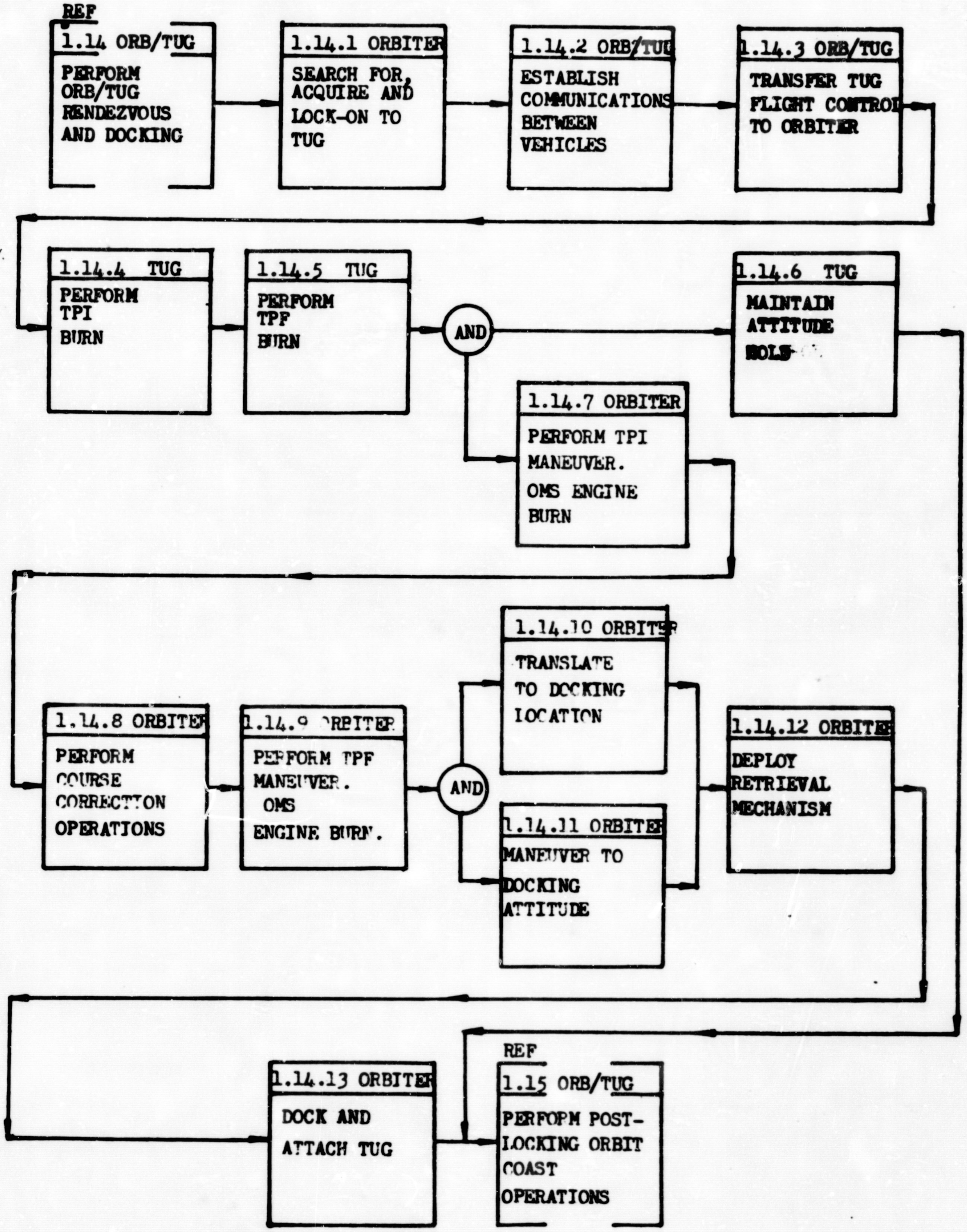

Figure A-13. . Second-Level Functions-Perform Orbiter/ Tug Rendezvous and Docking (All Missions)

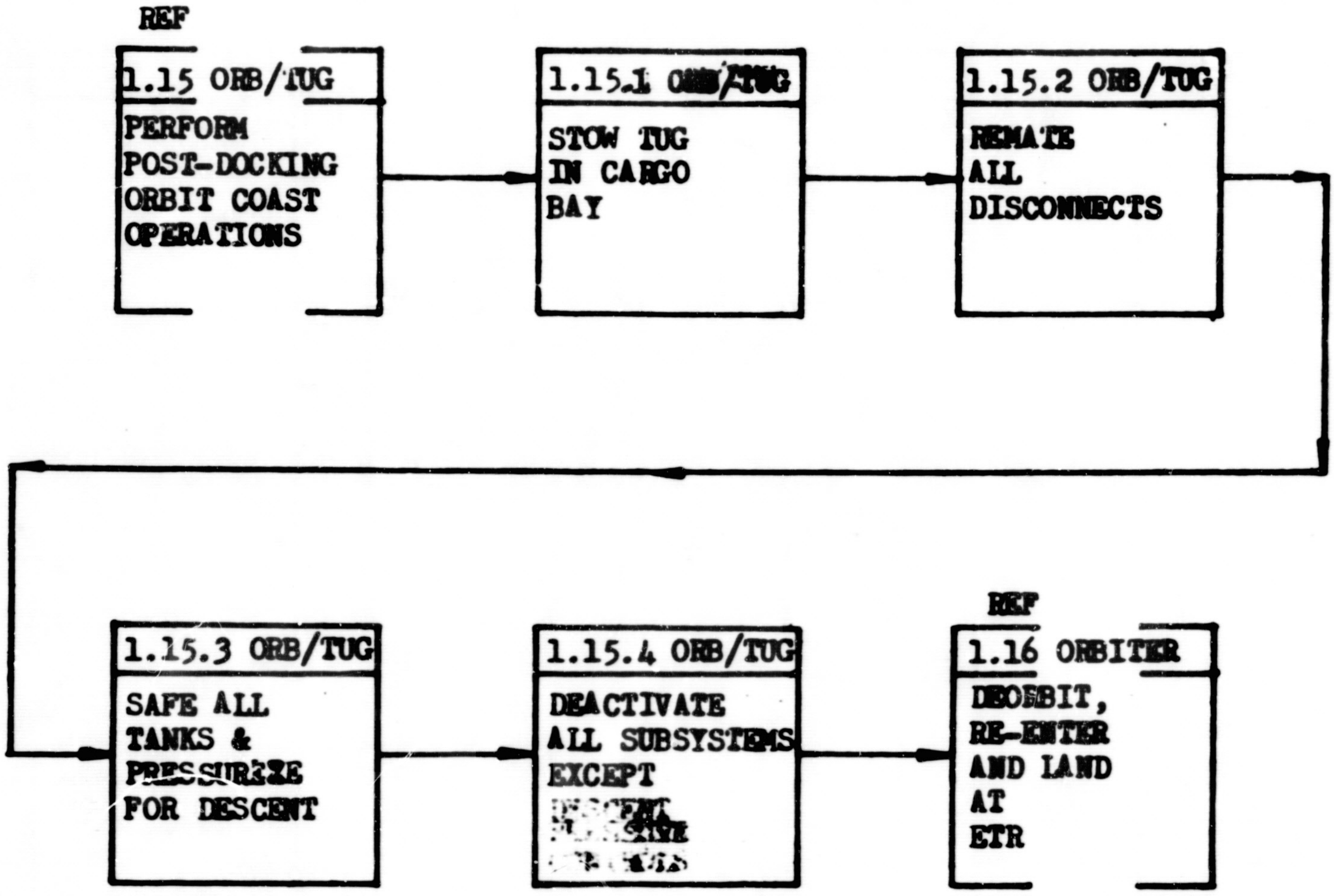

Figure A-14. SECOND-LEVEL FUNCTIONS - PERFORM POST-DOCKING ORBIT COAST OPERATIONS

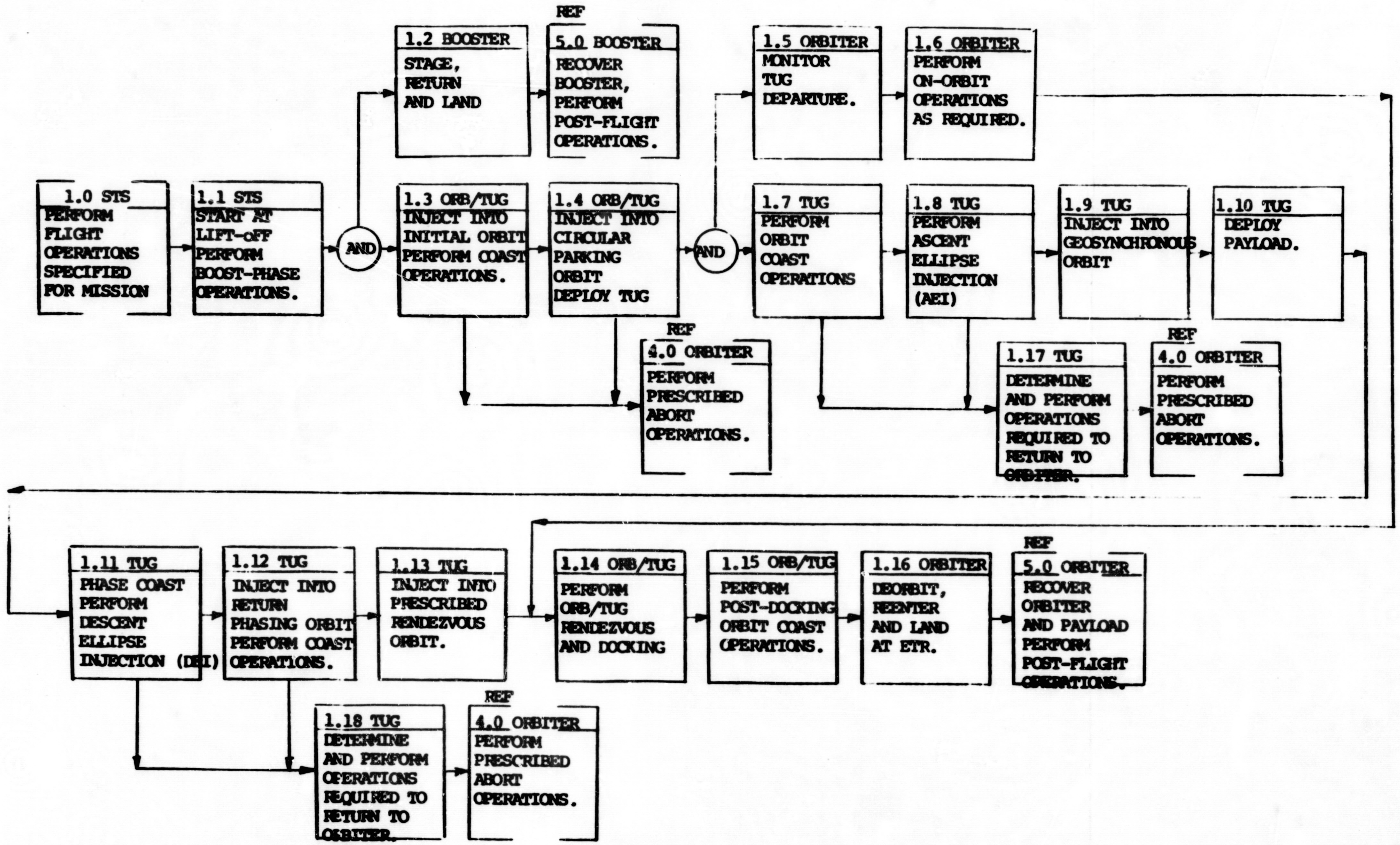

Figure A-15. First-Level Functions For Delivery Mission

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flow for the baseline mission except for the omission of the blocks (FFB's 1.19 and 1.20) associated with the retrieval of the return payload. This similarity does not necessarily imply that the design and operational impact resulting from the delivery mission is the same as the baseline mission. For example, the Tug/payload separation mechanism for the delivery mission could be much simpler since it is not required to dock with another payload for the return trip to the Shuttle. Similarly, ground operations involving test and checkout and the refurbishment of parts should also be simpler and the turn around time be shorter.

4.0 RETRIEVAL MISSION

The first level functions of the Retrieval Mission are shown in Figure A-16. The functions required for this mission are identical to the baseline mission except for the deletion of the payload deployment activities. The second level functions associated with each of the applicable first level blocks are the same for all missions. As previously mentioned, the similarity of a function does not mean that the design and/or operations impact is the same for every mission. Refer to the baseline mission discussion for explanation of the individual functional flow block as applicable to the Retrieval Mission.

5.0 GROUND AND SUPPORT FUNCTIONS ANALYSIS

5.1 Top and First-Level Functions

The top and first-level functions of the ground and support operations are sequentially depicted in Figure A-1 and Figure A-17, respectively. These functions are identified primarily for the baseline mission, but they should also be applicable to the retrieval mission as well as the delivery mission. The only difference is that on the retrieval mission the Tug is launched without a payload, whereas on the delivery mission the Tug is returned from space without a payload.

The life cycle of the Tug (20 missions) begins with the assembly, inspection, test and acceptance of the subsystem elements (FFB 10.0). Once the Tug is satisfactorily acceptance tested, it is then transported to the launch facility (FFB 9.0). All minor repairs and normal maintenance will be at the launch facility (FFB 8.0). Upon completion of the needed repairs and maintenance (for turn around cycles only) the Tug is mated to its payload and then the subsystem is mated to the booster/orbiter. The entire Space Transportation System (STS) is given a pre-launch readiness checkout and final countdown (FFB 2.0). At the end of each mission, the orbiter/Tug (with or without a return payload) is recovered and safed (FFB 5.0) prior to being transported and/or ferried to the launch facility (FFB 7.0) to be recycled for the next flight.

5.2 Second Level Functions

To provide in-depth visibility of the Tug/PL in the ground turn around cycle (FFB's 5.0, 7.0, 8.0 and 2.0) second level functions are identified and presented in Figures A-18, A-19, and A-20.

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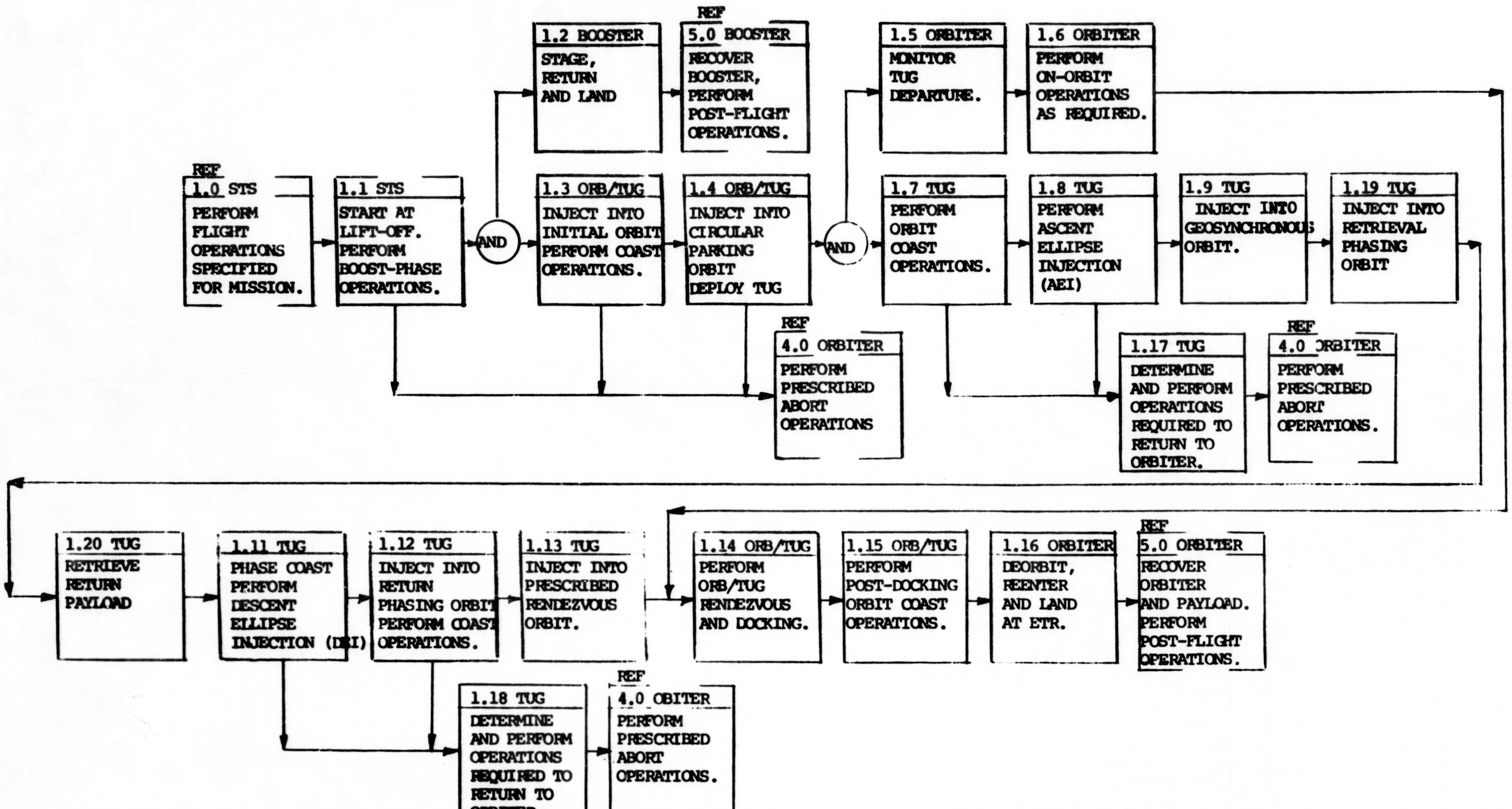

Figure A-16. First-Level Functions for Retrieval Mission

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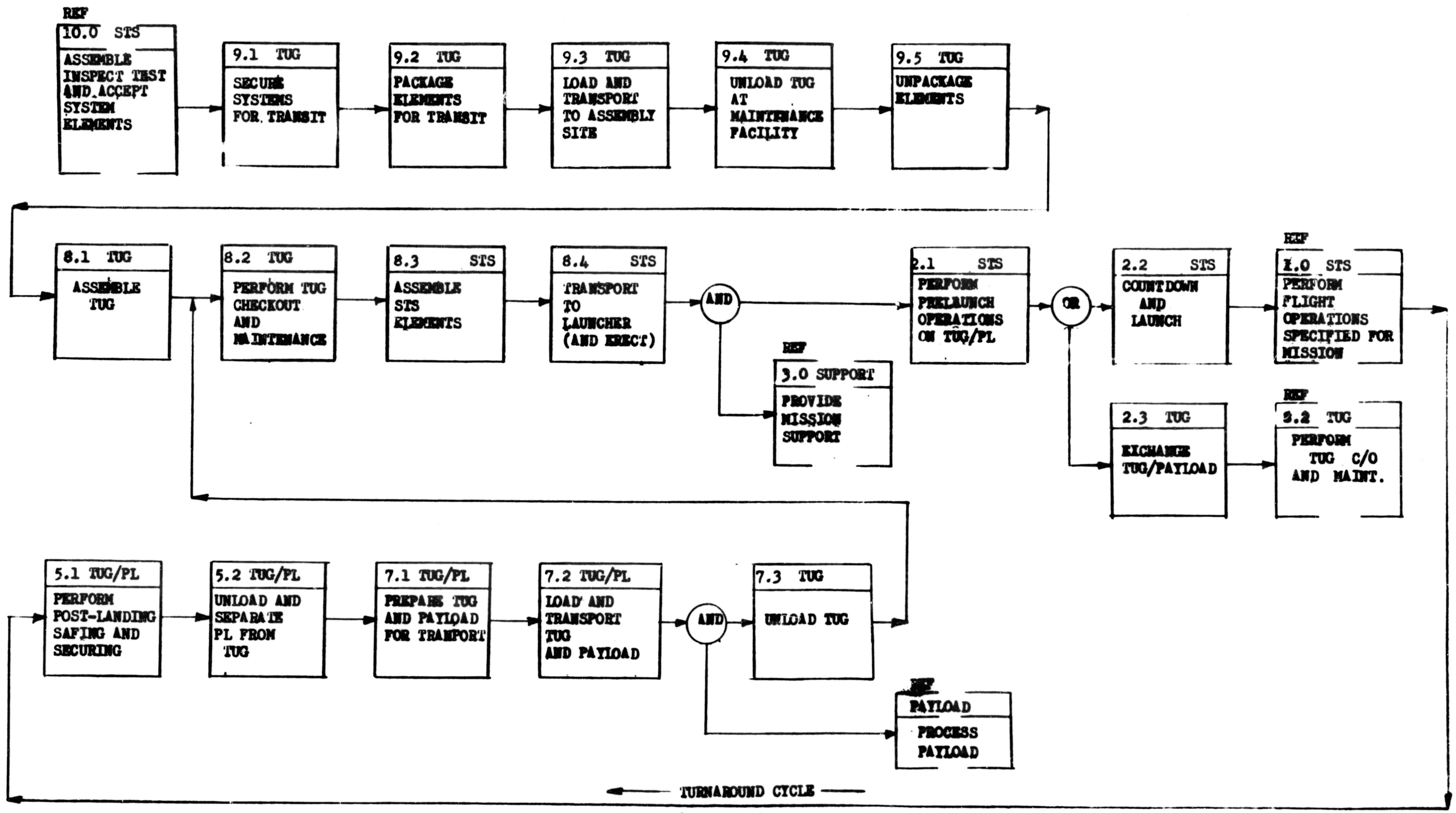

FIGURE A-17. FIRST LEVEL FUNCTIONS - TUG GROUND OPERATIONS

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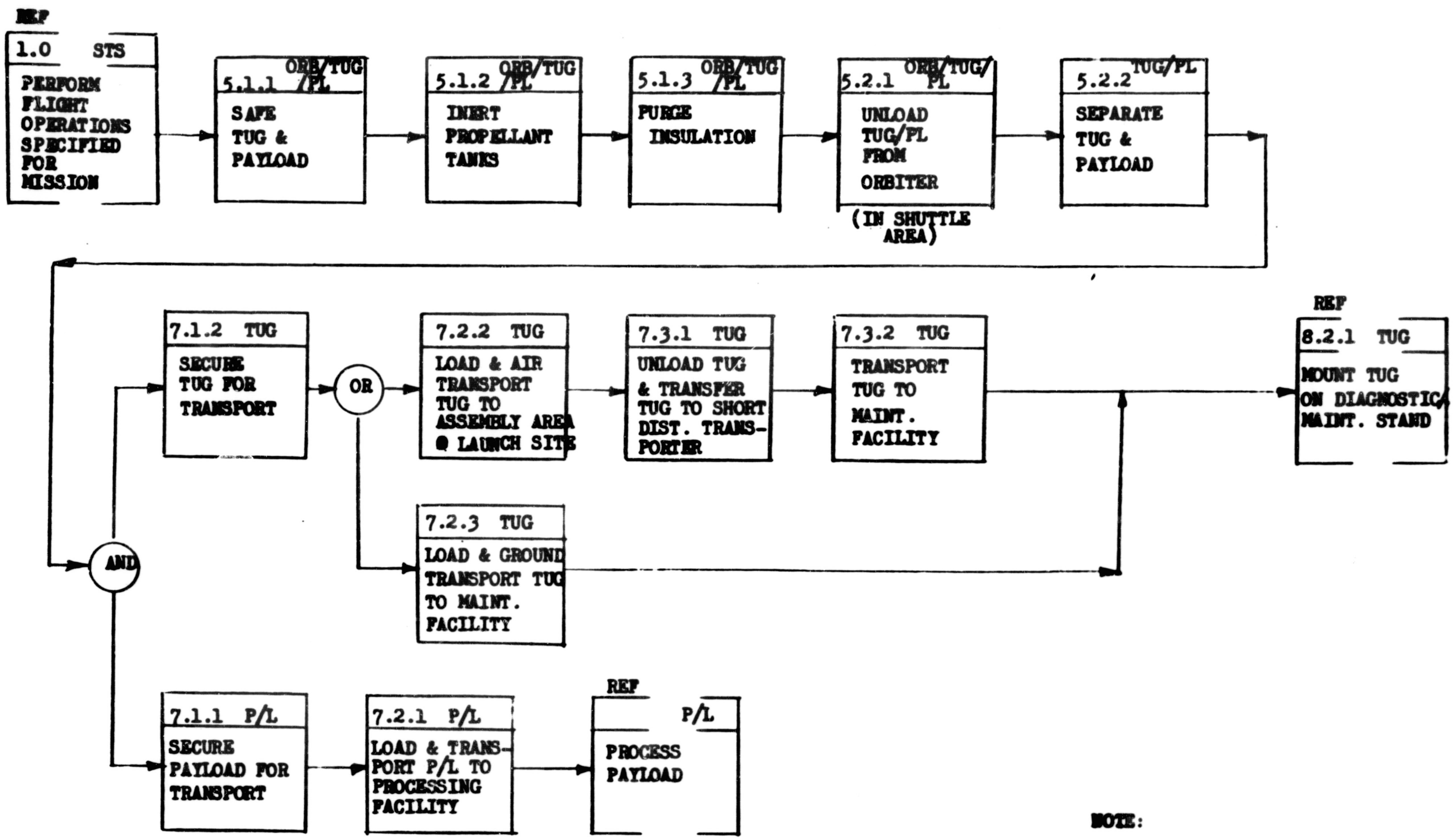

NOTE:

DELETE FFB'S 5.2-2, 7.1.1 AND 7.2.1 FOR DELIVERY MISSION.

FIGURE A-18. SECOND LEVEL FUNCTIONS - TUG GROUND OPERATIONS BASELINE AND RETRIEVAL MISSIONS

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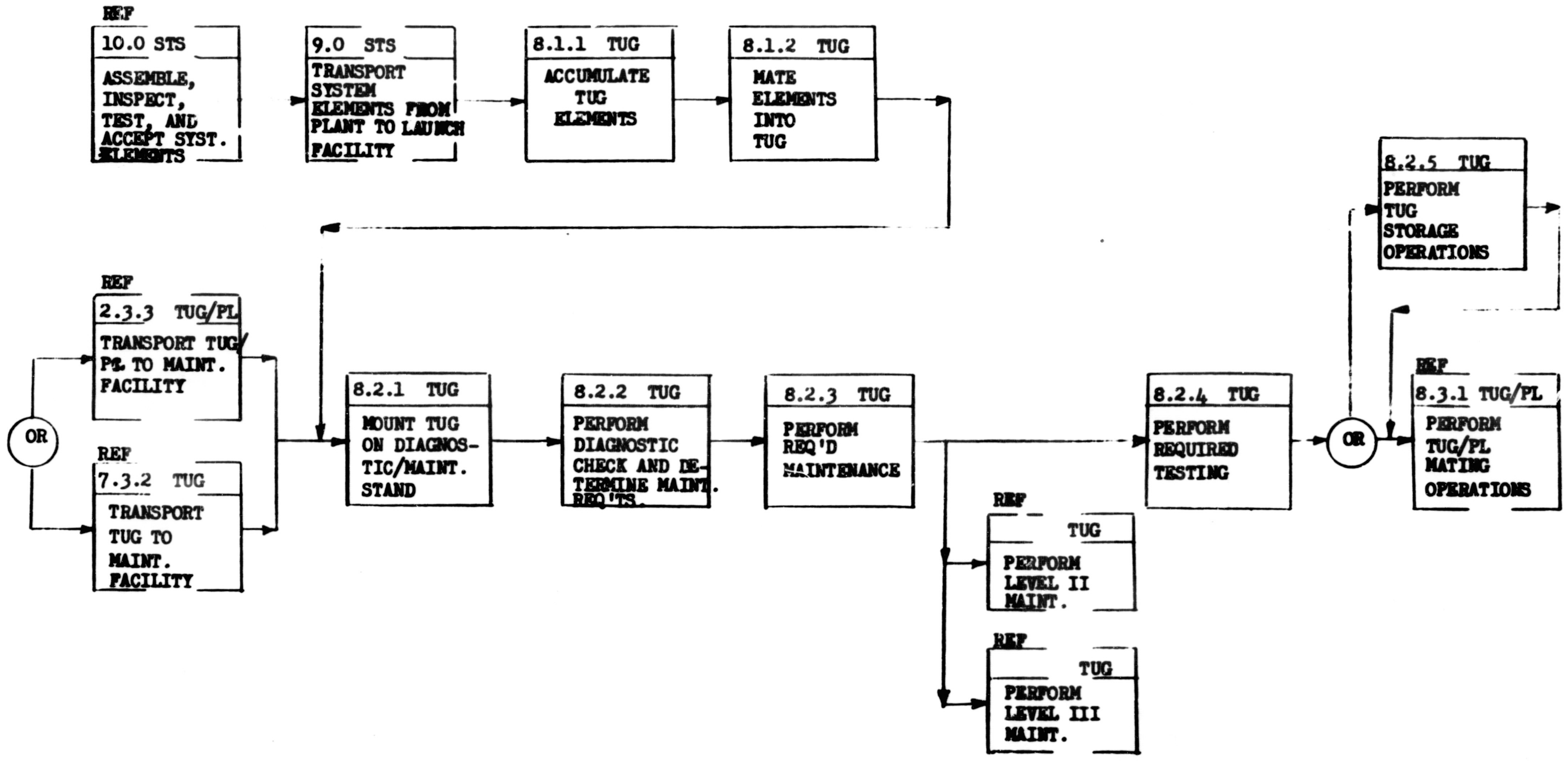

FIGURE A-19.

SECOND LEVEL FUNCTIONS - TUG GROUND OPERATIONS
ACCUMULATE TUG ELEMENTS

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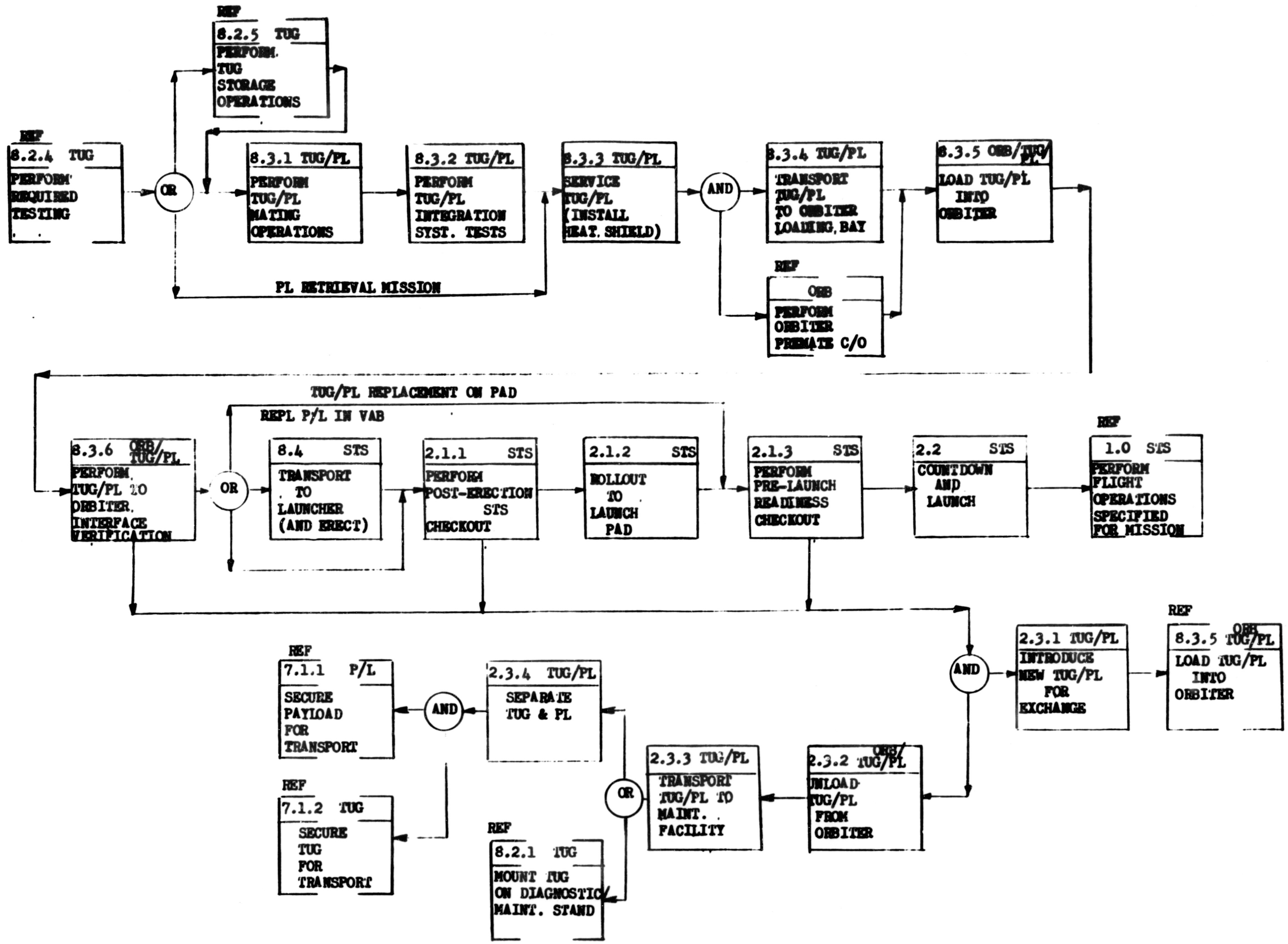

FIGURE A-20. SECOND-LEVEL FUNCTIONS - TUG/PL GROUND OPERATIONS PERFORM TUG/PL MATING OPERATIONS

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Upon return to earth, the Tug is safed (FFB 5.1.1), the propellant tanks inerted (FFB 5.1.2) and the insulation purged (FFB 5.1.3). If the return landing is made at the launch site, the orbiter is moved to the unloading area for the removal of the Tug/payload. However, if the landing is made at an alternate site, it may be necessary to airlift the unloading equipment to the site to facilitate the removal of the Tug/payload from the orbiter (FFB 5.2.1). Upon completion of the Tug/payload removal operation, the Tug is separated from its payload (FFB 5.2.2) and placed on separate transporters for shipment to the Tug maintenance facility (FFB's 7.1.2 through 7.3.2) and to the payload processing facility (FFB's 7.1.1 and 7.2.1). If the Tug is transported by air (FFB 7.2.2) it must be unloaded and transferred to another transporter (FFB 7.3.1) for the final trip to the maintenance facility (FFB 7.3.2).

After arriving at the maintenance facility, the Tug is mounted on the diagnostic/maintenance stand (FFB 8.2.1) and subjected to diagnostic checks (to determine subsystem performance status), and visual and nondestructive inspection. From the diagnostic check, corrective maintenance required to restore the Tug to its operating condition is identified (FFB 8.2.2). Under normal conditions, the Tug is serviced and maintained periodically (Level I maintenance) (FFB 8.2.3), and then tested to ensure that it will perform reliably (FFB 8.2.4). Level II maintenance is defined as maintenance consisting primarily of the modular replacement of components (or sub-assemblies). Certain elements (such as the fuel cell) are expected to be replaced periodically because of their expected MTBF. Level III maintenance consists of a major overhaul of the Tug. The main engine is expected to be replaced during this maintenance. Numerous other elements whose failure may be time related (i.e., seals) rather than cycle-based would also be replaced. Because of the long duration involved in the performance of the Level III maintanance, a separate maintenance area should be provided. This would permit the simultaneous performance of the Level I and Level II maintenance on the Tug replacement with a minimum disruption on the maintenance flow. Upon completion of the maintenance operations, regardless of level of maintenance, the Tug is then tested to verify the satisfactory accomplishment of the stipulated maintenance operations (FFB 8.2.4). From this point the Tug goes into storage operations (FFB 8.2.5) or it is mated with its payload (FFB 8.3.1) in preparation for the next flight. The integrity of the mating operations is verified both electronically and visually (FFB 8.3.2). The Tug/payload is then serviced and the heat shield installed (FFB 8.3.3). The heat shield referred to here is the protective shroud that protects the equipment located around the main engine nozzle from impingement by the exhaust plume. Upon completion of this operation, the Tug/payload is transported to the loading area (FFB 8.3.4) to be loaded into the cargo bay of the orbiter (FFB 8.3.5). The mated orbiter/Tug/payload is then given an interface verification test to ensure that all physical and electrical connections are properly mated (FFB 8.3.6). When this is completed, the entire subsystem is moved to a designated area for mating with the booster (FFB 8.4) and then given a post-erection test and checkout (FFB 2.1.1) prior to rollout to the launch pad (FFB 2.1.2). At the launch pad the entire Space Transportation System (STS) is given a pre-launch readiness checkout (FFB 2.1.3). If the STS passes the pre-launch checkout, it then goes into final countdown and lift-off (FFB 2.2).

If at any time, after the Tug/payload is loaded into cargo bay, a no-go situation occurs, the Tug/payload is removed from the orbiter and re-routed to the maintenance facility for further diagnostic checks. A new Tug/payload unit is introduced into the flow and mated with the orbiter (FFB 8.3.5) and the verification test sequence is repeated.

VOLUME II

APPENDIX B

DESIGN CRITERIA AND CONSTRAINTS DOCUMENT

CONTENTS

ILLUSTRATIONS

TABLES

ABBREVIATIONS AND ACRONYMS

ACS	Attitude Control System (ACPS and G&N)
ACPS	Attitude Control Propulsion System (Also RCS or APS)
APS	Auxiliary Propulsion System
AEI	Ascent Ellipse Insertion
APCU	Auxiliary Propulsion Conditioning Unit
BITE	Built-In Test Equipment
BSTR	Booster (Shuttle First Stage)
C/O	Checkout
COMM	Communication
CS	Communication Subsystem
db	Decibel
DMS	Data Management Subsystem
DEI	Descent Ellipse Insertion
DELTA-V	Velocity Change (ΔV) In Feet Per Second
DOD	Department of Defense
DRM	Design Reference Mission
ECS	Environmental Control Subsystem
EO	Earth Orbit
EPS	Electrical Power Subsystem
ETR	Eastern Test Range (Kennedy Space Center)
FFB	Functional Flow Block (Number)
FFBD	Functional Flow Block Diagram
FO	Fail Operational

FS	Fail Safe
GN&C	Guidance, Navigation, and Control
GH_2	Gaseous Hydrogen
GOX	Gaseous Oxygen (GO_2)
GN_2	Gaseous Nitrogen
GG	Gas Generator
h	Altitude (Nautical Miles)
h_a	Altitude of Apogee (N. Mi.)
h_p	Altitude of Perigee (N. Mi.)
IOC	Initial Operational Capability (Date)
IMU	Inertial Measuring Unit
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LOS	Line of Sight
LOX	Liquid Oxygen
LRU	Line Replacement Unit
MCC	Mid Course Correction
MLI	Multilayer Insulation
MPS	Main Propulsion Subsystem
MTTF	Mean Time to Failure
NPSH	Net Positive Suction Head
NPSP	Net Positive Suction Pressure
OBCO	On-Board Checkout
PC	Plane Change (Also P/C used)
POI	Phasing Orbit Insertion
RCS	Reaction Control System (Also ACPS or SPS)

SGLS	Space-Ground Link System (DOD)
SOI	Synchronous Orbit Insertion
SOW	Statement of Work
STS	Space Transportation System (Booster/Orbiter and Tug)
SV	State Vector
TOI	Transfer Orbit Insertion
TPF	Terminal Phase Final (Rendezvous Maneuver)
TPI	Terminal Phase Initial (Rendezvous Maneuver)
TVC	Thurst Vector Control
T/W	Thrust-to-Weight Ratio
λ	Tug Mass Fraction

INTRODUCTION

This document establishes the criteria to be used in planning the development and operation of the Space Tug. The design criteria and constraints were assembled and derived based upon mission operations, safety/reliability/maintainability, natural and induced environments to which the Tug will be exposed, functional requirements in meeting mission objectives, its interface with the Shuttle orbiter, payloads, facilities, and any constraints that may be imposed by manufacturing processes.

MISSION CRITERIA

MISSION CHARACTERISTICS

The Space Tug missions and their characteristics are defined in Table B-1.

The Tug will be designed for ground based operations with all propellant loading, payload/Tug assembly, maintenance, repair, and refurbishment to be accomplished on the ground. The Tug will be designed for a 20 mission life with multiple burn engine vacuum starts and propellant pressure cycles limited to 160 cycles each.

Tub orbital operations include Tug undocking and redocing with Shuttle orbiter and payload, minimum functional test of Tug prior to its separation from the Shuttle orbiter and delivery of the payload to and from its target orbit.

Design Reference Mission

The basic mission requirement for Tug is to place and/or retrieve payloads at various inclinations in high Earth orbits, including geosynchronous orbits. The Space Tug baseline design mission is to transport an unmanned 3,000 pound payloads from low earth orbit (100 N.M. 28.5° inclination) to geosynchronous orbit and return an equal payload to low Earth orbit. The payload/Tug combination will be shuttled between earth surface and the low earth orbit by the Shuttle orbiter.

The baseline mission flight profile of Space Tug is illustrated in Figure B-1 with mission segments indicated by numbered events on the mission profile and identified in Table B-2. Major mission phases can be identified as: Boost to 100 N.M. operations orbit, ascent to synchronous orbit, payload placement/retrieval, descent to 270 N.M. phasing orbit, descent to 100 N.M. operations orbit and deorbit to landing. The Space Tug will be launched to the circular operations orbit (100 N.M., i = 28.5°) in the Space Shuttle orbiter cargo bay. This mission phase includes events 1 through 5 as identified on the mission profile of Figure B-1. Events 6 and 7, Tug checkout and orbit coast,

Table B-1. Space Tug Design Mission Characteristics

CHARACTERISTICS \ MISSION TYPES	BASELINE GEOSYNCHRONOUS PAYLOAD EXCHANGE	GEOSYNCHRONOUS PAYLOAD DELIVERY	GEOSYNCHRONOUS PAYLOAD RETRIEVAL
Departure Orbit Altitude/Inclination	100 N. Mi./28.5° Inclin.	100 N.Mi./28.5°	100 N.Mi./28.5°
Target Orbit Altitude/Inclination Longitude	19,300/0° TBD	19,300/0° TBD	19,300/0° TBD
Return Phasing Orbit Altitude/Inclination	270/28.5°	270/28.5°	270/28.5°
Rendezvous Orbit Altitude/Inclination	100/28.5° (within 300 meters of Shuttle orbiter)	100/28.5° (within 300 meters of Shuttle orbiter)	100/28.5° (within 300 meters of Shuttle orbiter)
Duration	6 days on orbit + 1 day in Shuttle Orbiter	3 days on orbit	3 days on orbit
Payload Weight CG at Geometric Center of 15' X 25' P/L Envelope	3,000 Lbs	8,060 Lbs	4,160 Lbs
Notes	Retrieve 3,000 lb. P/L Within 6,000 N.Mi. of Delivered P/L.	Zero P/L Return	Zero P/L Delivery

respectively, lead to the next major mission phase of ascent to geosynchronous orbit bounded by events 8 (burn to synchronous orbit injection) and 10 (circularize in synchronous orbit). The placement/retrieval phases of the mission are bounded by events 11 (deploy payload) and 16 (phasing orbit coast). During these phases, the Tug will place a 3,000 pound payload at its designated longitude, rendezvous with and engage a 3,000 pound return payload that is within 6,000 N.M. of the delivered payload. Alternate Tug payload mixes include the delivery of an 8,060-pound payload to synchronous orbit with zero payload return to the operations orbit or retrieval of a 4160 pound payload from synchronous orbit with on payload delivery to synchronous orbit as indicated in Table B-1.

The descent to 270 N.M. phasing orbit is initiated by event 17 with the engine retro burn and is terminated by the circularization burn event 19. Duration of the phase on-orbit coast may vary from zero to twenty-two hours before the next mission phase "Descent to 100 N.M. Operations Orbit" is initiated. In the operations orbit, the Tug executes the terminal rendezvous to within 300 meters of the Shuttle orbiter, where upon it assumes an attitude hold position for final engagement by the Shuttle orbiter. Following completion of stowing the Tug in the orbiter cargo bay, the last phase of the mission is completed with deorbit and landing of the Shuttle Orbiter and Tug/payload combination.

Tug mission duration (lift-off to landing) can vary from 0.93 to 6.48 days, depending upon the need for orbit phasing as indicated in the flight profile of Figure B-1 and the timeline of Table B-2. The Tug is to be designed for an on-orbit stay time of six days detached from the Shuttle orbiter. It also will be designed to stay one additional day inside the Shuttle cargo bay in a standby condition during Earth-to-orbit ascent and orbit-to-earth descent. After landing, the Tug will be capable of remaining in the Shuttle Orbiter cargo bay in a safe condition for a maximum of 24 hours prior to Tug removal to the maintenance/refurbishment area.

FLIGHT CRITERIA

a. For purposes of performance comparison calculations, all Tug missions will be initiated from and terminated at the Shuttle orbiter in a 100 N.M. circular orbit inclined at 28.5 degrees.

b. The Tug payload shall be considered as self-sustaining with the Tug requirement for placement and retrieval in or from the target orbit.

c. A communications satellite is assumed to be available if required for Tug use.

d. Continuous communications and tracking is not required.

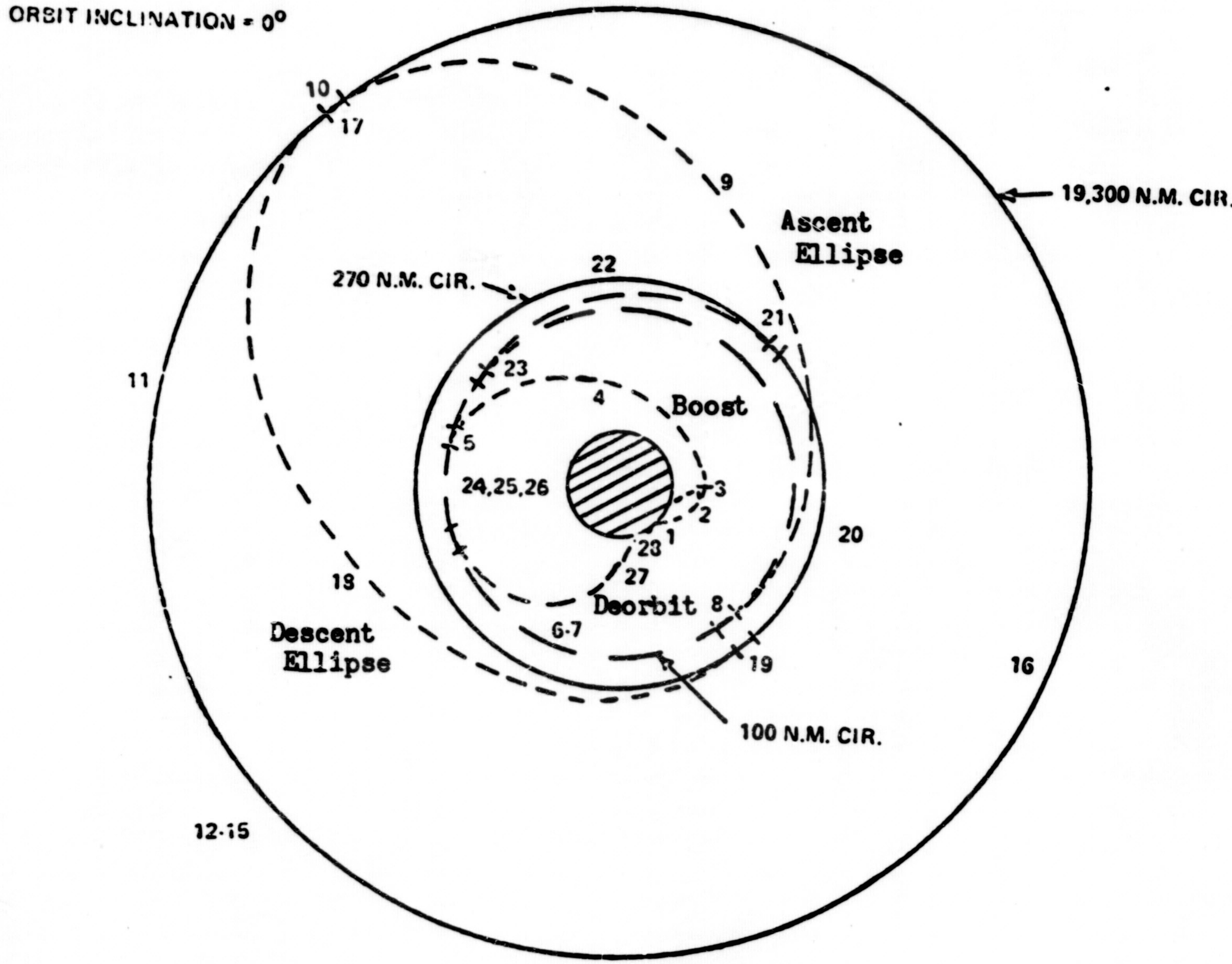


Figure B-1 Baseline Mission Profile and Event Sequence

Table B-2 Tug Timeline (Liftoff to Landing)

TIME SEQUENCE (Days)	(Hours)	EVENTS Duration (Hours)	No.		TUG DELTA-V (FPS)	RCS
0.0	0.00	0.05	1	FIRST STAGE BOOST		
	0.05	0.00	2	BOOSTER/ORBITER STAGING		
	0.12	0.07	3	INJECT INTO 50 BY 100 N.Mi., 28.5° INCLINED ORBIT		
	0.85	0.73	4	COAST TO 100 N.Mi. APOGEE		
		0.00	5	CIRCULARIZE SHUTTLE ORBITER		
	2.85	2.00	6	DEPLOY TUG/PAYLOAD		
	14.85	12.00	7	PHASE IN ORBIT FOR LONGITUDE (0-12 HRS)		
		0.00	8	INJECT INTO 100 BY 19,300 N.Mi., TRANSFER ORBIT -2° PLANE CHANGE	8,446	
0.84	20.13	5.28	9	COAST TO 19,300 N.MI, APOGEE		
		0.00	10	CIRCULARIZE AND CHANGE PLANE 26.5°	5,893	
	21.13	1.00	11	DEPLOAY PAYLOAD IN ORBIT		30
3.88	93.13	72.00	12	PHASE IN ORBIT TO RETRIEVE RETURN PAYLOAD (0-72 HRS)	100	
	95.13	2.00	13	RENDEZVOUS WITH PAYLOAD	100	15
4.01	96.13	1.00	14	DOCK TUG AND PAYLOAD		
	96.63	0.50	15	SAFE PAYLOAD		
4.53	108.63	12.00	16	PHASE IN ORBIT TO RETURN TO SHUTTLE ORBITER (0-12 HRS)		
		0.00	17	INJECT INTO 19,300 BY 270 N.MI. TRANSFER ORBIT - PLANE CHANGE 26.5°	5,721	
4.74	113.95	5.32	18	COAST TO 270 N.MI. PERIGEE		50
		0.00	19	CIRCULARIZE AND PLANE CHANGE 2° TO 270 N.MI., 28.5° INC.ORBIT	7,867	
5.66	135.95	22.00	20	PHASE IN ORBIT TO RETURN TO SHUTTLE ORBITER (0-22 HRS)		
		0.00	21	INJECT INTO 270 BY 100 N.MI. TRANSFER ORBIT	296	
5.70	136.71	0.76	22	COAST TO 100 N.MI. PERIGEE		
		0.00	23	CIRCULARIZE IN 100 N.MI. ORBIT	296	
5.78	138.71	2.00	24	RENDEZVOUS TUG AND SHUTTLE ORBITER	100	15
5.82	139.71	1.00	25	DOCK TUG AND SHUTTLE ORBITER		10
6.45	154.71	15.00	26	PHASE IN ORBIT TO RETURN TO LAUNCH SITE (0-15 HRS)		
		0.00	27	DEORBIT FOR RETURN TO EARTH		
6.48	155.41	.70	28	LAND AT LAUNCH SITE (42 MIN)		

e. Communications requirements for Tug operations are:

To Tug	From Tug
Control (P/L Dock)	System Telemetry
Computer Data	Computer Data
Tracking	T.V.
Nav. Data	

f. Tug is the active vehicle in rendezvous and docking with passive payloads. Terminal Tug/payload docking accomplished with man-in-the-loop television.

g. Tug will rendezvous to within 300 meters of Shuttle orbiter and maintain stabilized attitude.

h. Tug docking to the payload and Shuttle Orbiter shall meet the docking accuracy requirements specified in Table B-3.

Table B-3. Docking Accuracy Requirements

Parameter	Structural	G&C
Centering Miss Distance	0 to 1.0 Foot	0 to 0.75 Foot
Miss Angle	0 to 5.0 Degrees	0 to 1.0 Degree
Longitudinal Velocity	0.1 to 1.0 Ft/Sec	0 to 1.0 Ft/Sec
Lateral Velocity	0 to 0.30 Ft/Sec	0 to 0.3 Ft/Sec
Angular Velocity (Combined maximum of pitch, yaw, & roll motion)	0 to 0.50 Deg/Sec	0 to 0.50 Deg/Sec

i. Tug will not provide any payload pointing or maneuvering services while on orbit except those required to transport the payload to or from its desired orbit.

j. Deployment of the Tug payload in its desired orbit shall leave the payload in a stable mode and within the following limits: Position TBD, Velocity TBD, and Attitude Rate TBD.

k. All Tug propellant dumping, system safing and inerting will be done with the Tug in the Shuttle Orbiter cargo bay. Propellant settling for this function will be provided by the Shuttle Orbiter. Tug propellant will be dumped and gas vented at the Shuttle Orbiter skin through Tug/Shuttle Orbiter umbilicals.

1. Tug propellant tank safing is defined by the following sequence of operations: (1) Dump propellant to space vacuum (1 psia), (2) Pressurize propellant tanks to 17 psia with helium gas, (3) Vent helium/ propellant gas mixture to space vacuum (1 psia), (4) Repressurize propellant tanks with helium gas to 17 psia and hold this pressure throughout deorbit and landing.

RELIABILITY/SAFETY CRITERIA

RELIABILITY

The objective of the Tug Reliability/Safety effort is to assure the design development and manufacture of a vehicle in which minimum hazard potential and maximum probability of mission success are inherent characteristics.

Tug Criticality Definitions are listed below:

CRIT IA - Any single point (first order) failure which will result in injury or loss of the Orbiter crew.

CRIT IB - Any single point (first order) failure which will result in physical destruction of payload.

CRIT IIA - Any single point (first order) failure in the Tug which will result in loss of the payload mission objectives.

CRIT IIB - Any single point (first order) failure which will result in unsafe condition for the Orbiter crew and vehicle or which will cause loss of Tug mission objectives.

CRIT III - All others.

SAFETY

Preliminary safety criteria for ground, ascent/descent, and mission operations are listed below and shall be incorporated in the Tug design. Safety criteria are categorized as they apply to all systems or individual systems.

All Systems

1. Inadvertent activation of critical systems shall be precluded through design considerations and the use of protective devices.

2. The capability shall be provided for performing critical functions (CRIT IA and IB) at a nominal level with any single component failed.

3. Provisions shall be made to eliminate metallic or non-metallic particles or parts, which could cause failure through lack of protection of the electrical equipment, in a zero "g" environment.

4. Provisions shall be made to protect equipment from damage or corrosion caused by electrolyte leakage from batteries.

5. The use of sulphur-containing or sulphur-coated materials in close proximity to electrical contacts is prohibited.

6. Adjacent connections, electrical or fluid, should be configured differently to minimize cross connections.

Propellant System

1. No Shuttle cargo should be permitted to leak, vent, or discharge propellants into the cargo bay.

2. Avoid venting bulk quantities of liquids if solid particles/chunks could impede subsequent operations.

3. The design of liquid dump lines must take into account that liquids discharging into a vacuum will evaporatively solidify and may restrict or block liquid flow.

4. A Tug which has sustained impact and penetration of propellant tanks by meteoroids or space debris should not be loaded into the cargo bay of the Shuttle until it can be verified that no liquid propellants remain.

5. The purge capability for inerting of integrated propellant systems will encompass the propellant system being inerted, from source through vent exhaust.

6. Automatic protective equipment will be incorporated in vehicle cryogenic systems to sense the onset of ullage pressure drop during the initial period of cryogenic loading, i.e., pressure level drops which could result in system failure (collapsing of tanks or tubing) due to cryogenics.

7. Cryogenic subsystems shall be designed to preclude the exposure of electrical/electronic components to cryogenic fluids.

8. Cryogenic fluid fill and drain systems shall be designed to minimize sensitivity to two-phase flow conditions and associated phenomena such as geysering.

Pressurization System

1. Structures designed only for positive pressure will have provisions for vacuum relief and for the prevention of inadvertent depressurization.

2. **Bypass circuits which override system pressurization interlocks are prohibited.**

Electrical System

1. Electrical components cooled to cryogenic temperatures shall be designed to preclude the ingestion of moisture and salts from ambient environments.

2. Relays shall be design oriented along the sensitive axis to minimize the effects of vibration with redundant relays oriented along orthogonal axes.

3. Leads of electronic parts shall be configured in a way that provides for relief of strain due to thermal expansion and contraction.

Structural System

1. Venting, drainage, and disposal provisions will be designed into structural compartments for the safe elimination of hazardous fluids.

2. Vent and drain openings at the vehicle mold line, including equipment drains, will be located such that drainage or vented gases will not re-enter the vehicle through other drain holes or vents.

3. All structural and mechanical systems will be designed with materials which are compatible with all working fluids and flushing and cleaning agents under the stress and environmental conditions of use.

4. Insulation will be compatible with the fluid within the propellant system to avoid hazards in the event of leaks.

ENVIRONMENTAL CRITERIA

NATURAL ENVIRONMENT

Terrestrial climatic - NASA TMX 53872, "Terrestrial Environment (Climatic) Criteria Guidelines for Use in Space Vehicle Development," 1969 revision, shall be used as the model for climatic conditions the Tug will be exposed to at the fabrication, test, and launch locations. The Tug shall be designed to withstand these environments during test at the site. Protective equipment shall be provided for the Tug during transportation and storage to protect it from environmental conditions in which it was not intended to operate.

Space Environment - NASA TMX 53957, "Space Environmental Criteria Guidelines for Use in Space Vehicle Development," dated September 1969, shall be utilized as the environmental model for terrestrial space. Subject matter covered in the document include the following: definition of area of influence, meteoroid environment, radiation environment (solar high energy particle) properties, radiation (thermal), magnetic field, wind regimes, ionosphere, solar particle prediction, and gravitational data.

INDUCED ENVIRONMENT

In addition to the natural environments discussed in the preceding paragraphs, the Tug will be subjected to induced environments caused by boost to orbit and return to earth in the Space Shuttle orbiter cargo bay. The Tug will be designed to be carried to orbit and returned from orbit with or without the payload attached to the Tug in the Space Shuttle and must be designed to meet the following Shuttle environmental characteristics. All environmental definitions are defined as those present at the interface of the Shuttle and the cargo bay extremities. Any amplification factors contributed by Shuttle/Shuttle payload interface mechanisms must be accounted for in addition to those shown below. The three payloads mixes identified in Table B-1 must be taken into consideration in the Tug design.

Loads

The loads imparted to the Shuttle cargo are defined in Table B-4.

Table B-4. Shuttle Payload Load Factors

Condition	X(g)	Y(g)	Z(g)
Launch	1.4 ±1.6	±1.0	±1.0
High Q Booster Thrust High Q Booster Thrust	1.9 ±0.3	±1.0	0.8 ±0.2
End Boost (Booster Thrust)*	3±0.3	±0.6	±0.6
End Burn (Orbiter Thrust)	3±0.3	±0.5	±0.5
Orbiter Entry	-0.5	±1.0	-3.0 ±1.0
Orbiter Flyback	-0.5	±1.0	+1.0 -2.5 ±1.0
Landing	-1.3	±0.5	-2.7 ±0.5

*Excludes booster-orbiter separation loads which are TBD

Thermal

The Shuttle Orbiter cargo bay internal wall temperature thermal environment is defined in Table B-5.

Table B-5. Temperature Limits for the Internal Walls of the Cargo Bay

Condition	Temperatures (°F)	
	Minimum	Maximum
Pre-launch	-100	+120
Launch	-100	+200
On-orbit (door closed)	-100	+200
On-orbit (door open)	--	--
Entry and post landing	-100	+200

During ascent, while attached to Shuttle in orbit and during re-entry, the Shuttle Orbiter shall not provide thermal control for Tug or Tug payload systems.

Acoustic

The Shuttle Orbiter cargo bay acoustic environment is defined in Table B-6.

Vibrations

(1) Vehicle Dynamics
Longitudinal axis (3-35 Hz at 3 Oct/Min) TBD
Lateral axis (3-35 Hz at 3 Oct/Min) TBD

(2) Liftoff Random Vibration
20-2000 Hz TBD
Time TBD

(3) Boost Random Vibration
20-2000 Hz TBD
Time TBD

(4) Shock Spectrum
20-100 Hz TBD

Table B-6. Orbiter Payload Compartment Internal Acoustic Design Criteria (Sound Pressure Level (db) Ref. $10^{-5} N/M^2$)

1/3 Octave Center Band Freq. (HZ)	Lift-Off	Boundary Layer
5	124	124.5
6.3	127	125.0
8	128	126.0
10	129	126.5
12.5	131	127.0
16	132	128.0
20	134	128.5
25	135	129.0
31.5	137	130.0
40	138	130.5
50	139	131.0
63	140	132.0
80	141	132.5
100	143	133.0
125	144	134.0
160	145	134.5
200	145	135.5
250	145	136.0
315	144	136.5
400	143	137.0
500	142	137.5
630	141	138.0
800	140	138.5
1K	139	138.0
1.25K	138	137.0
1.6K	137	136.5
2K	135	135.5

Table B-6. Orbiter Payload Compartment Internal Acoustic Design Criteria (Sound Pressure Level (db) Ref. $10^{-5}N/M^2$) (Cont)

1/3 Octave Center Band Freq. (HZ)	Lift-Off	Boundary Layer
2.5K	134	134.5
3.15K	133	134.0
4K	132	133.0
5K	131	132.0
6.3K	130	131.0
8K	129	130.0
10K	128	129.0
	OASPL155db	OASPL149db

Payload Bay Atmosphere

The Shuttle Orbiter payload bay is capable of atmospheric control independent of Shuttle Orbiter internal structure while on the launch pad. The payload bay shall be vented during launch and entry phase and operate unpressurized during the orbital phase of the mission. Cargo bay pressure is assumed to equal the orbiter ambient pressure during all operational phases. The cargo bay will be inerted with N^2 on ground and during ascent to orbit.

VEHICLE DESIGN CRITERIA

GENERAL

Configuration

The Tug is designed as a single stage orbital propulsion vehicle with a mass fraction target of 0.895 or better. The targeted mass fraction includes a 10 percent weight factor reserved for Tug growth only. Figure B-2 represents the baseline Tug inboard profile configuration. It has a maximum overall diameter of 15 feet and a maximum length of 35 feet (including Shuttle Orbiter/Tug and payload/Tug docking mechanisms). It is designed to operate with or without a payload. Maximum payload is limited to a diameter of 15 feet and a length of 25 feet. The Tug accommodates a single high performance LOX/LH_2 main engine. The avionics equipment is mounted on the skirt forward of the LH_2 tank, which has hemispherical bulkheads, and the RCS engines are mounted toward the aft end on the outer structure. Tug/Shuttle Orbiter and Tug/payload interface points are located as indicated in Figure B-2. The docking systems should be designed such that the active portion is left with the Shuttle Orbiter in the Tug/Shuttle Orbiter docking interface and with the Tug in the Tug/payload docking interface. The Tug/Shuttle Orbiter docking system is to be designed to carry lateral and axial loads from the Tug and payload to the Shuttle during ascent and reentry. The docking systems should be designed for maximum Tug performance and need not necessarily be identical systems. As indicated in Figure B-3, the Tug will be oriented in an inverted attitude during ground launch of the Space Shuttle which will necessitate certain Tug subsystems be capable of operation in an inverted attitude under varying gravity conditions of launch and boost. Figure B-3 presents the loads sign convention to be utilized in the loads analysis and Figure B-4 presents the axis sign convention and coordinates to be utilized in control analysis.

Redundancy

The Tug will be designed as a fail-safe vehicle which has no failure mode which could cause the payload to be destroyed or a mode which could cause an unsafe situation for the Space Shuttle or its crew.

a. Subsystems except primary structure and pressure vessels shall be designed to fail safe after the failure of the most critical component to preclude destruction of the payload or danger to the Space Shuttle or its crew.

b. In systems where redundancy is needed, the Tug systems shall be developed only to provide limited mission capability to safeguard payload, Space Shuttle, and Space Shuttle crew.

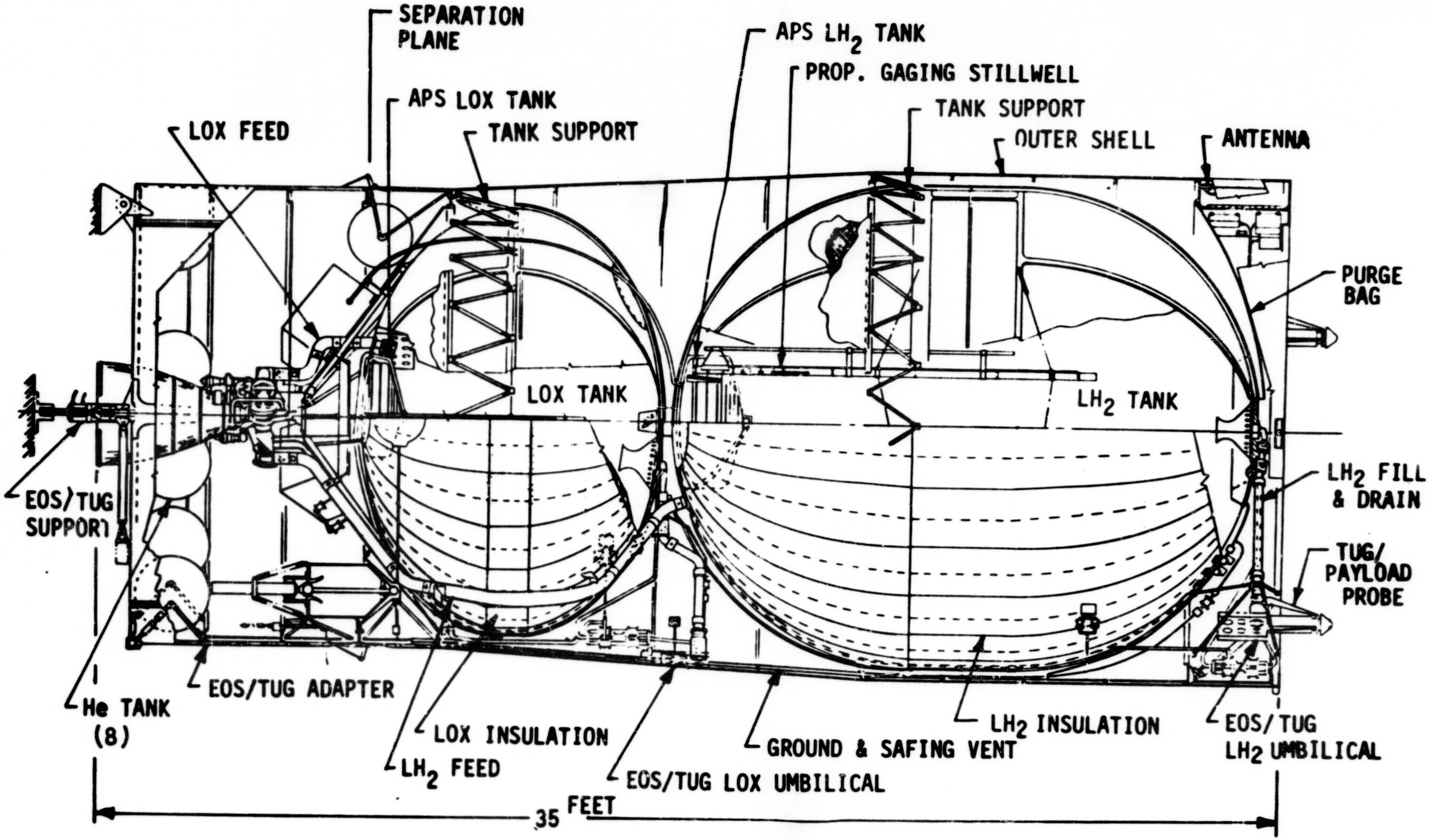


Figure B-2 Tug Point Design Inboard Profile (Sheet 1 of 2)

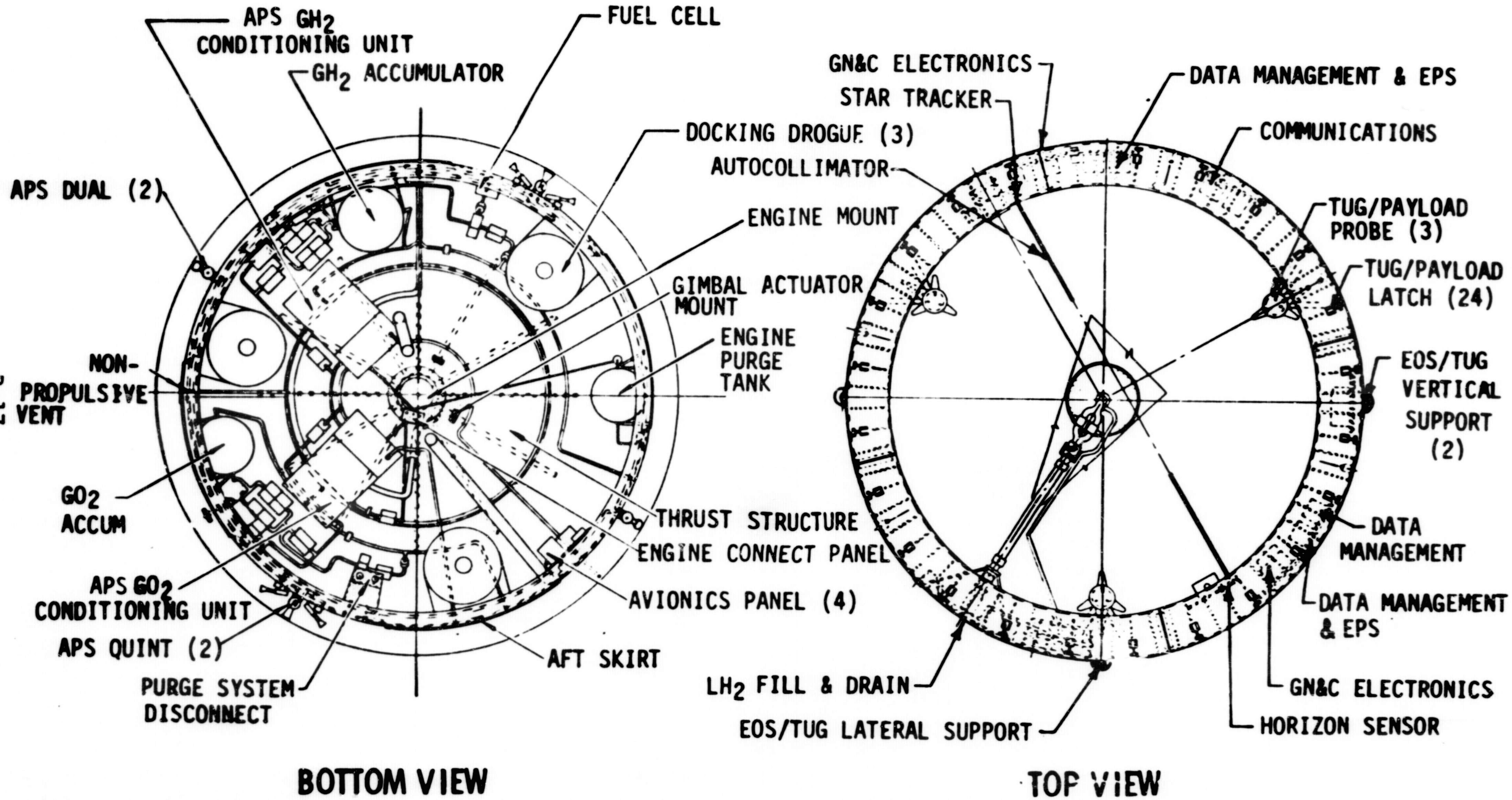


Figure B-2 Tug Point Design Inboard Profile (Sheet 2 of 2)

When Integrating Nose to Tail

+ Force Produces + Shear
+Y Shear Produces + Bending Moment About Z
+Z Shear Produces - Bending Moment About Y

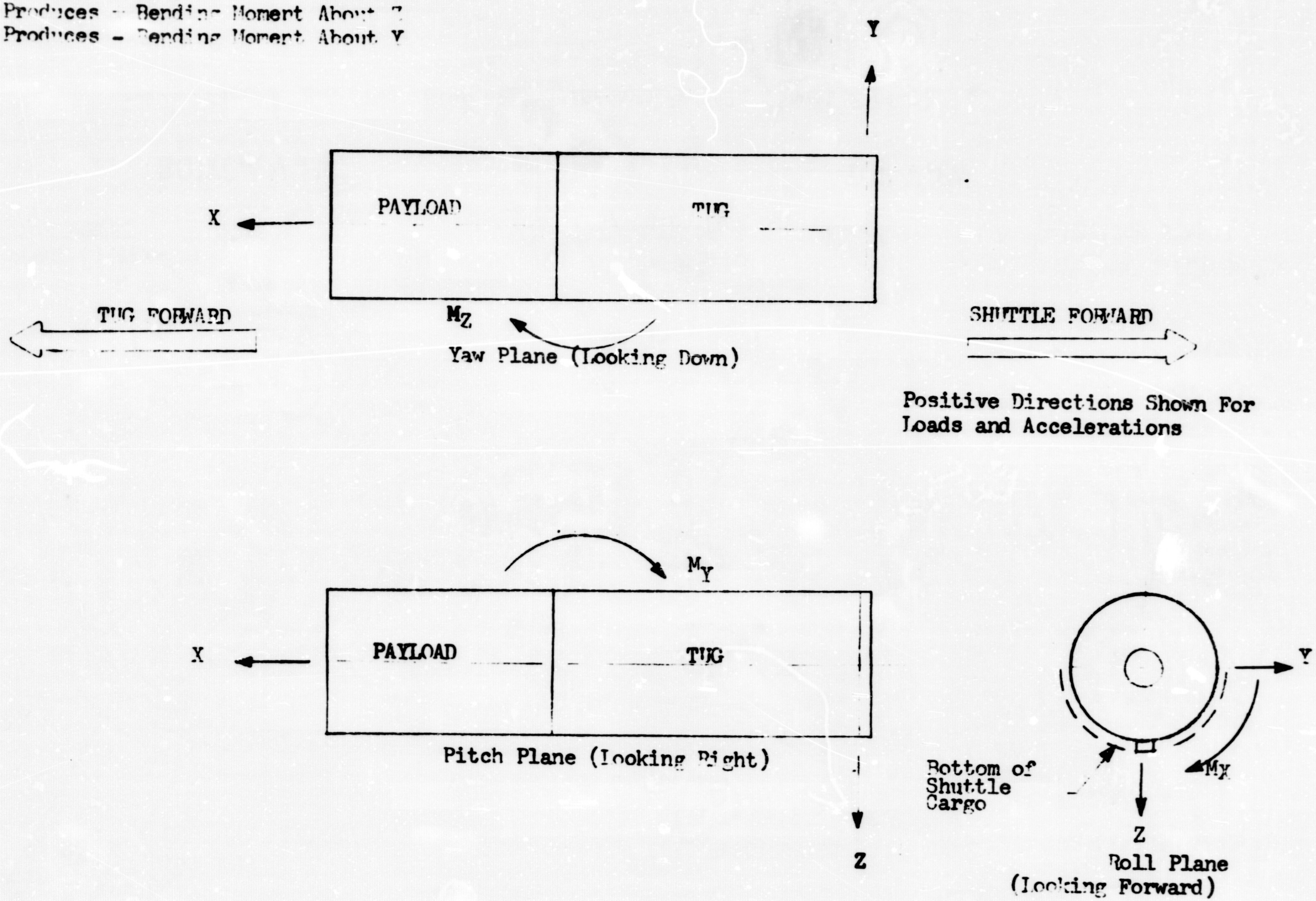


Figure B-3. LOADS SIGN CONVENTION

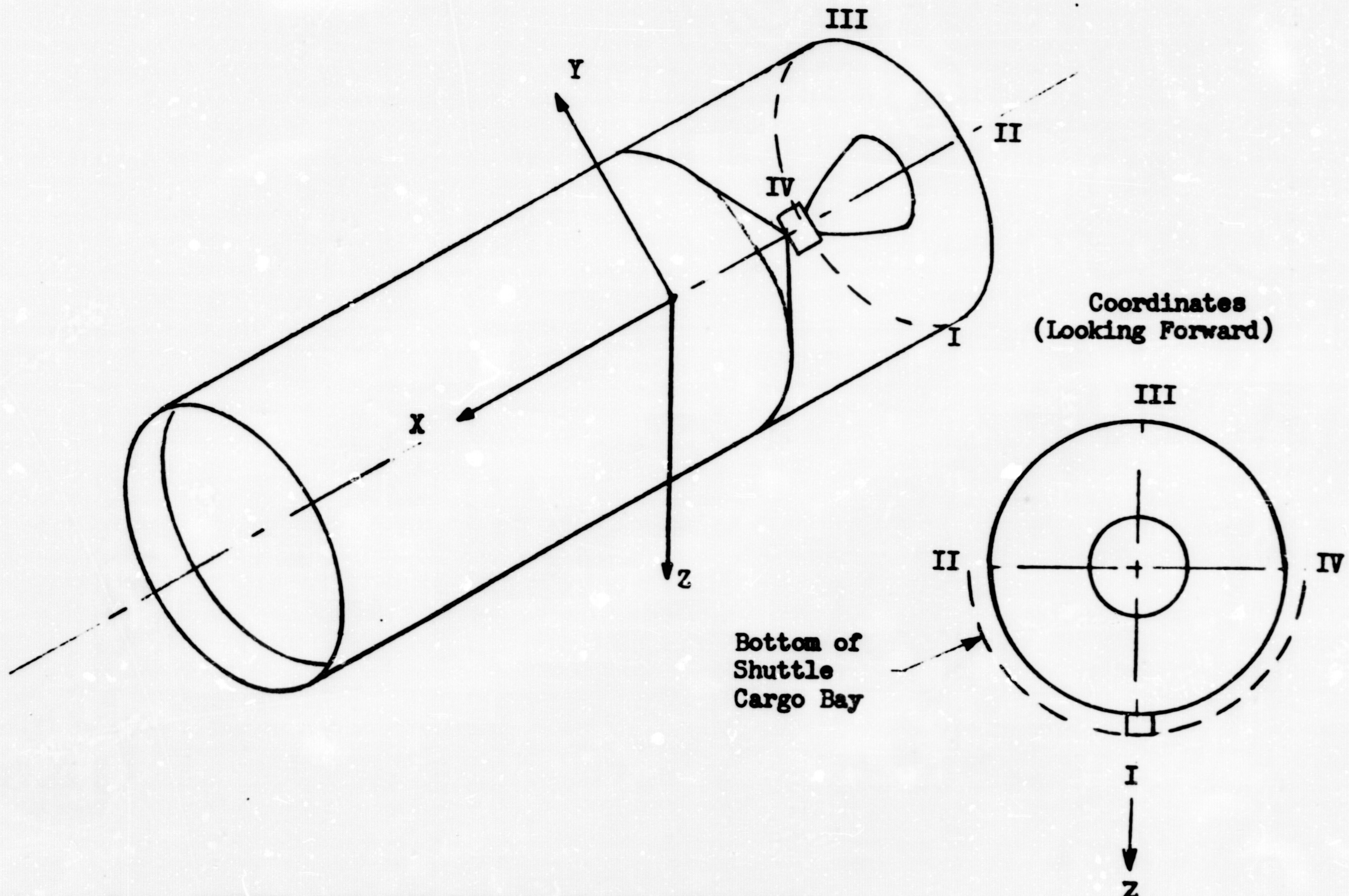


Figure B-4. CONTROL AXIS SIGN CONVENTION AND COORDINATES

c. In addition to built-in redundancy, alternate modes of operation are acceptable to fulfill redundancy requirements when their use does not create the need for additional hardware.

d. Redundant components susceptible to similar shock, vibration and acceleration loads shall be physically oriented such that they will not fail from the same causes.

e. Alternate or redundant capability shall be provided in the ACPS to enable the Tug to hold attitude to permit recovery of Tug for a period of 30 minutes following a failure in the primary system.

f. The up-link and down-link portions of the Telemetry System should be redundant to insure proper control of the Tug for all conditions.

g. Back-up low weight rate gyro system or equivalent is required for attitude hold mode.

h. Back-up emergency battery will supply loads for 30 minutes.

Testability/Inspectability

The Tug is to be designed for ground-based operations with all maintenance, repair, and refurbishment to be done on the ground. In order to enable these operations to be accomplished effectively, the following design guides shall be observed to facilitate test and inspection with due concern for the vital need to minimize Tug inert weight:

a. The design of vehicle subsystems incorporating redundancies shall include a means of verifying satisfactory operation of each redundant path at any time the system is determined to require testing.

b. All critical systems or circuits, whose malfunction could result in unsafe or potentially hazardous situations, shall be monitored. Intelligence of the condition will be provided to ground stations or Shuttle crew so that corrective action can be initiated.

c. Automatic Tug control systems which are critical to payload and Space Shuttle/crew safety shall be provided with the capability for remote manual override.

d. Electrical and fluid subsystems shall include checkout test points which will permit normal planned system checkout tests to be made without disconnecting tubing or electrical connections which are normally connected in flight.

e. A Tug design objective wherever feasible without severe weight impact shall be to meet the basic requirement of accessibility, maintainability, repairability and inspection:

(1) Equipment shall be located and fastened so that it can be reached easily for rapid inspection, testing, servicing, removal and replacement.

(2) There shall be ample in-position space in access area for inspection and maintenance operations on components and installation.

(3) Consideration will be given to implement the latest development in Nondestructive Test (NDT) techniques in the design to facilitate Tug inspection and testing. An example of NDT current technology now permits access provisions from 3/8" to 1/2" to perform inspections that previously required man-entrance. Table B-7 lists the non-destructive technology and techniques that should be considered in the Tug design and establish provisions to permit their accomplishment. Any one or combination of the NDT applications may be utilized to enable the effective accomplishment of inspection and testing during fabrication, flight cycle or maintenance cycle.

(4) Where possible, design to allow component disassembly without invalidating previous testing or adjustment.

f. Procurement and certification specifications for items such as seals, quick disconnects, and mechanical components shall require environmental exposure cycle demonstration test requirements to at least twice design operational life.

g. The following testing provisions shall be considered:

(1) Give high priority to provision for rapid trouble shooting by use of go-no-go type of test equipment.

(2) Plan for use of quantitative test to check the operation of an overall subsystem for making minor adjustments to bring operation within prescribed limits.

(3) Test connections and points must be located on front of components and modules.

h. Subsystems that are intended to operate in zero or multiple g environments shall, wherever practicable, be capable of test and verification in one g environment during ground maintenance.

i. The subsystem design shall consider means to activate or deactivate components without disturbing other systems. For example, electronic line replaceable unit (LRU) replacement should not require deactivation of other electronic systems.

Materials

Material selection and properties will be based on projected 1976 technology with due consideration given to materials compatibility for space application.

a. Titanium, magnesium, lead or any of their alloys shall not be used where exposed to liquid oxygen nor where exposed to gaseous oxygen exceeding two atmospheres of absolute pressure.

Table B-7. Nondestructive Technologies

Nondestructive Technologies	Techniques
1. Ultrasonics	Pulse echo, through transmission, resonant, reflector
2. Visual	10X maignification, flexible fiber optics, rod optics, black light (ultraviolet)
3. Radiographic	Conventional X-ray, neutron, isotope, simulated electron emission, Beta backscatter
4. Acoustic Emission	Sonic analysis, flaw growth detection
5. Holographic Interferometry	Pulsed/impulse load, real time/thermal load vibration time average, double exposure
6. Thermography, Emissivity	Liquid crystals, scanning radiometer, infrared image conversions
7. Eddy Current	Flaw detection, thickness measuring, material properties
8. Microwave	Attenuative, scatter, reflective
9. Penetrants	Fluor[illegible]nt
10. Magnetic Particle	Dry powder, wet method
11. RF Testing	Dielectric constant, capacitance, dielectric strength, leakage current
12. Thermal Indicating Paints	Single temperature measurement
13. Leak Test	Helium, ultrasonic detection
14. Fatigue Gages	Bondable foil gage responds to fatigue loading by increase in resistance readout in microhms
15. Photography	Color, infrared, and ultraviolet

b. The risk of galvanic corrosion in flight vehicles shall be minimized by consideration of relative electrical potential (E.M.F.) in the selection and application of metals. Metals that differ in potential by more than 0.25 volt shall not be used in direct contact when exposed to a common electrolyte such as the atmosphere.

c. Zinc or cadmium plating shall not be used in the Tug systems.

d. Insulation and other materials will be compatible with the fluid within the fluid system to avoid hazards in the event of leaks. The material will be non-wicking and should not react with moisture, propellants, or other agents to produce corrosive, toxic, flammable, noxious or otherwise undesirable by-products.

e. Components which contain mercury shall not be specified for use with systems or their testing, checkout, handling, or maintenance equipment.

f. Polyvinylchloride (PVC) shall not be used in Tug systems.

g. Plated fittings, joints or plumbing shall not be used in oxygen systems.

Maintainability

The Tug will be designed for an operational life of 20 missions and refurbishment of subsystems, except for structure and tankage, after each mission is acceptable. Structure and tankage shall be designed to meet the 20 mission operational life with no need for scheduled maintenance whereas scheduled maintenance of other subsystems will be determined by mean time to failure of components in the systems. Maintainability is a design characteristic which permits servicing, checkout, fault detection and isolation, repair, replacement, and refurbishment of the Tug systems, subsystems and components in an efficient and cost effective manner. Two prime factors driving Tug maintainability design characteristics are targeted Tug mass fraction (≥ 0.895) and ground based maintenance operations. The Tug will be designed as an integral vehicle,with none of the subsystems designed to be removable as a kit or single unit, in order to minimize Tug weight. Subsystem replacement as a unit is not required since the maintenance operations are ground based as opposed to spaced based.

Appropriate consideration of maintainability criteria will be reflected in the design development of the Tug vehicle and support equipment. Criteria provided herein are general guidelines to be considered for incorporation in the Tug concept development, item selection, design trade-offs, and design reviews. Specific criteria will be developed for subsystems and component application as the design definition progresses.

General

The following general guidelines will be observed in all design approaches wherever possible consistent with the urgent need to minimize weight:

a. Reduce the complexity of maintenance by:

1. Providing adequate accessibility, work space and clearance.

2. Providing for interchangeability of like components, materials, and parts within the system.

3. Utilizing standard parts and items within the existing industry and government inventories.

4. Limiting the number and variety of tools, accessories and support equipment.

5. Insuring compatibility among this system and, Shuttle equipment and facilities.

b. Reduce the need for and frequency of design-dictated maintenance activities by using:

1. Components which require little or no scheduled maintenance for the 20 mission life of the Tug.

2. Tolerances which allow for wear throughout life of part.

3. Adequate corrosion prevention and control features.

c. Reduce the maintenance task times by designing for:

1. Detection of malfunction or degradation to replaceable unit level.

2. Ease of fault correction.

3. On-board verification of corrective action.

4. Rapid and positive adjustment and calibration.

5. The reduction of human error caused by incorrect assembly, connection, or installation.

6. Tank entry ports in propellant tanks for maintenance and inspection.

Installation Criteria

1. Consider the effects of vertical and horizontal vehicle orientation on servicing checkout and maintenance requirements.

2. Optimize arrangements and accessibility of systems and components in relationship to frequency and priority of replacement and nature of maintenance function to be accomplished.

3. Design for maintenance work in a standing position. Allow space for change of posture if prolonged kneeling, bending, or crouching is necessary.

4. Adequare illumination should be provided in closed areas where frequent maintenance activity will occur.

5. The hinge pin, in pin-type doors, must be readily accessible and easily removable.

6. Design to avoid requiring excess personnel in a work area.

7. Location of critical components and systems must consider the number of maintenance and/or modification personnel which may be assigned to work in the area.

8. Doors and access panels should allow for use in either vertical or horizontal positions.

9. Mark all access doors for major subsystem function if possible.

10. Locate individual servicing provisions and areas for easy and simultaneous access for both equipment and personnel. Consider horizontal and vertical orientation.

11. Design for component installation to be accomplished in a single-direction mounting maneuver. Avoid the need for complex or stepped directional changes to insert the component into its mounted position.

12. Provide supports, guides, and guide pins to assist in aligning and positioning of units.

13. Avoid use of shimmed adjustment installations.

14. Provide footholds, handholds, steps, and nonskid surfaces where necessary.

15. Equipment access, location, mounting, and power supply shall not expose personnel to hazards such as electrical shock and burns. Where necessary, guards shall be installed to protect personnel.

16. Fluid lines should be marked for fluid type, pressure range, and direction of flow. Electrical wiring should be identified with wire number, which includes system identification, wire size and potential.

17. Quick change capability for all GSE interface fittings shall be provided.

18. Component configuration and mounting must allow for minimum replacement time.

19. Enclosed maintenance areas such as avionics bay must be painted white for better visibility.

20. Fluid lines, cables, and ducting should be routed to avoid sharp bends, protuberances and moving surfaces that could wear away material, causing leaks or failure.

21. Where possible, cables, fluid lines, and ducting must be routed through vehicle structure to avoid being used for handholds or steps.

22. Fluid drains should be configured to avoid impingement of the fluids on any part of the vehicles and to avoid any entrance into any openings. Vehicle mated positions must be considered; e.g., the Tug should not drain hydraulic fluid into orbiter cargo bay.

23. Design should allow ground servicing operations to be performed simultaneously.

24. Consideration should be given to installation of GSE power and communications circuits (to remote locations) on the vehicles. These provisions will facilitate maintenance action when power or communication is required for verification and safety.

25. Avoid bending loads on GSE interface fittings by heavy umbilicals.

Installation and Subsystem Criteria

1. Installation must allow for fast and easy adjustments.

2. Access considerations must make allowances for the individual to see, feel, and manipulate components.

3. The subsystem design shall consider means to activate or deactivate components without disturbing other systems. For example, avionics LRU replacement should not require deactivation of other electronic systems.

4. Only a minimum number of latches should be used on panels where frequent access is required.

5. Doors should be of appropriate size consistent with the functions served, the dimensions of the unit involved, and the maintenance action required.

6. All doors and access covers requiring seals should have provisions for rapid seal replacement.

7. Unique fittings should be used to avoid mixing of incompatible fluids.

8. Design of fittings and connections should preclude dirt and dust contamination, and leakage when disconnected.

9. Provide clearance to all mounting provisions to enable use of standard hand tools for component removal.

10. All access openings, doors, and panels should be standardized to two or three sizes.

11. The following items should be considered for design as replaceable and interchangeable items: all plastic/fiberglass ducting, all access panels with quick change fasteners, all access panels attached with screws, thermal protective system.

12. Maintenance placards are recommended for system schematics, servicing requirements and detail maintenance information when used frequently.

13. Maintenance warning information should utilize red background placards.

14. Avoid sequential maintenance tasks whenever possible. Alternate methods should be considered to avoid potential compounded delay situations; e.g., if systems or component maintenance task in first of sequenced tasks is not completed in time allowable, the second sequenced task may become the cause of serious delay should the second task also exceed time allowable.

15. Installation should allow for adequate in-position inspection and repair capability of components and systems.

16. Design for ease and speed of detection and isolation of faults. Fault indication should be recorded or retained by latching indicators.

17. Functional check capability should be maximized to expedite fault isolation and to enhance system monitoring.

18. Specific subsystem test points should be grouped at one point and clearly identified to facilitate checkout.

19. Where possible, automatic test equipment connectors should be provided on LRU's for ease of ground checkout after LRU removal.

20. Positive design shall be provided to prevent reverse installation of components.

21. Consideration must be given to maintenance by replacement vs trouble-shooting to minimize elapsed time. Unit must be easy to remove and must not require post-installation functional check adjustment. Checks to verify operation must be rapidly accomplished and extensive full functional checks after installation should be avoided.

22. Maximize use of LRU's to enhance fault isolation and decrease system restoration time.

23. Plan reference points in the rigging concept to permit engine controls rigging by use of electronic or optical sighting techniques as opposed to the need for mechanical gauges and rigging boards.

24. The preferred installation for splined cantilevered components is to use a slotted (keyhole) mounting system, whereby the component is inserted and rotated to lock into position. Final securing then may be accomplished by using a single bolt clamp device (V-band, Quad-ring, etc.).

25. Within the reasonable design constraints for transportability, minimum disassembly for shipment is required to reduce maintenance manpower and to reduce error.

26. Mechanical retention devices for LRU's should not require safety wiring.

27. Sensitive adjustments should be protected by guards, covers or locking devices.

28. Aluminum threads should be avoided.

29. Design should allow removal of subcomponents from a subassembly wherever practical without requiring removal of the whole subassembly.

30. Use quick detach modular component detachment including bearings and pillow blocks; avoid use of pressed bearings in vehicle structure.

31. Size and weight of components should be controlled to practical man handling limits. When size and weight must exceed these limits, hard points for hoisting should be provided.

32. Methods and equipment to permit checkout without requiring removal of equipment from the vehicle must be developed simultaneously with system/component design.

33. All ground operations must be accomplished without the need to clear people or equipment from adjacent areas or the need to take special precautions to protect equipment. (Does not include prelaunch or launch operation.)

34. Provision for nondestructive inspection (NDI) of critical structure and components should allow fast and easy installation of NDI sensors and transmitters.

35. Use standard connections and design to use standard servicing equipment for fluids and gases.

Structural Materials

Structural material selection and properties will be based upon projected 1976 materials technology; projected materials properties are subject to MSFC review.

Environmental Criteria

Load factors applied to the Space Tug and payload while being transported in the Space Shuttle cargo bay will be those valves specified in Table B-4. Tug operation including Tug/payload and Tug/Shuttle docking will not exceed load factor valves applied to the Space Tug and payload specified in Table B-4. Vibration and acoustic environment specified in Table B-6 will be applied to the Space Tug and payload while being transported in the Space Shuttle. Determination of the probability of micrometeoroid penetration of the Space Tug will be made utilizing the micrometeoroid environments specified in NASA TMX-53957 "Space Environment Criteria Guidelines for use in Space Vehicle Development" (1969 revision) 2nd edition, August 26, 1970. Probability of no penetration shall not be less than 0.95 during the 20 mission service life.

Transporting and Ground Handling

Transportability of the Tug shall require minimum disassembly or configuration change for transporting the vehicle. Tug design will provide for hard points to facilitate handling and transportation. Handling and transportation equipment will be designed such that no structural load conditions exceeding the flight environment are encountered. Where these conditions cannot be met, Tug packaging will be designed to absorb loads that exceed the defined flight loads.

Protective covers will be provided to protect the Tug from climatic elements during shipment from the fabricating site to the launch site. Moist air will be excluded from the propellant tanks, insulation, and equipment by means which will not impose weight penalties on the stage design. This may be accomplished by GSE supplied purge or maintaining positive pressure to exclude the detrimental climatic element.

The prime mode of transportation between the fabrication site and launch site will be by Guppy or other commercial air line. GSE will be provided to preclude tank collapse resulting from changing atmospheric pressures encountered in air transportation. A positive tank pressure is to be maintained in the propellant tanks throughout the entire transportation period.

The Tug and payload will be handled as an integral unit when being installed in or removed from the Shuttle Orbiter cargo bay. Installation of the Tug alone in the Shuttle Orbiter will occur in the payload retrieval missions and conversely the Tug alone will be unloaded from the Orbiter cargo bay in the payload delivery missions. All Tug handling installation or removal operations will be accomplished with propellant tanks empty. In the event a Tug is returned with propellant in its tank(s), the propellant tank(s) will be drained and/or vented prior to removal of the Tug from the Orbiter cargo bay.

STRUCTURE

The Tug structure will be designed to meet the following criteria: Safety Factors - All primary and secondary structural components where critical load conditions occur while in the Space Tug is attached to the Space Shuttle will be designed with minimum allowable safety factors of 1.4 ultimate and 1.1 yield. For a structural component whose critical load condition occurs during Tug operations or other time, where failure of the component will have no effect on the Space Shuttle System, the component may be designed with minimum allowable safety factors of 1.25 ultimate and 1.05 yield. All structural components will be designed for positive margins of safety for 20 mission cycles.

Propellant Tankage

The Space Tug propellant tankage will be designed employing fracture mechanics specified in NASA-SP-8040 "Fracture Control of Metallic Pressure Vessels" based upon mission temperature/pressure time profile. The temperature/pressure data used for fracture mechanics evaluation of the Tug LOX and LH_2 tanks is listed in Table B-8. The mission temperature/pressure history for the LOX and LH_2 tanks are presented in Figures B-5 and B-6 respectively.

Table B-8. Tug Temperature/Pressure Data

Phase	No. of Cycles	Pressure PSIG LH2	Pressure PSIG LOX	Temperatue °F LH2	Temperatue °F LOX
Proof Test	(1)	26.2	30.8	Room Temp.	Room Temp.
	(1)	33.6	35.0	-423	-320
		Pressure PSIA LH2	Pressure PSIA LOX	Temperature °F LH2	Temperature °F LOX
Static Firing					
Pretest Checkout	(1)	24	24	Room Temp.	Room Temp.
Test	(3)	24	24	-423	-279
Post Test Checkout	(1)	24	24	Room Temp.	Room Temp.
Flight					
Preflight Checkout	(20)	24	24	Room Temp.	Room Temp.
Flight	(20 missions as defined in Figures B-5 and B-6.				
Post-Test Checkout	(19)	24	24	Room Temp.	Room Temp.

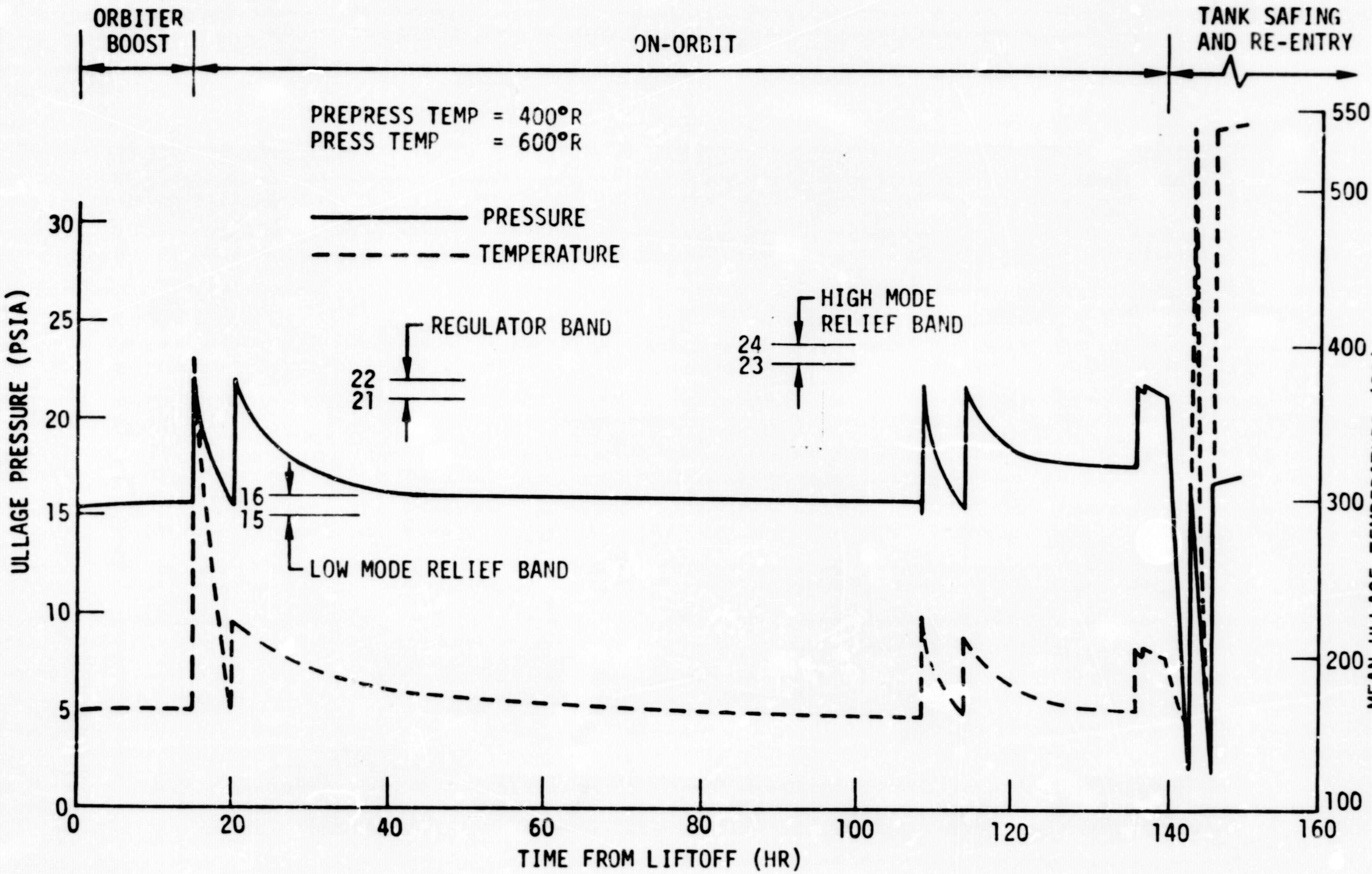


Figure B-5. Main LOX Tank Temperature/Pressure History

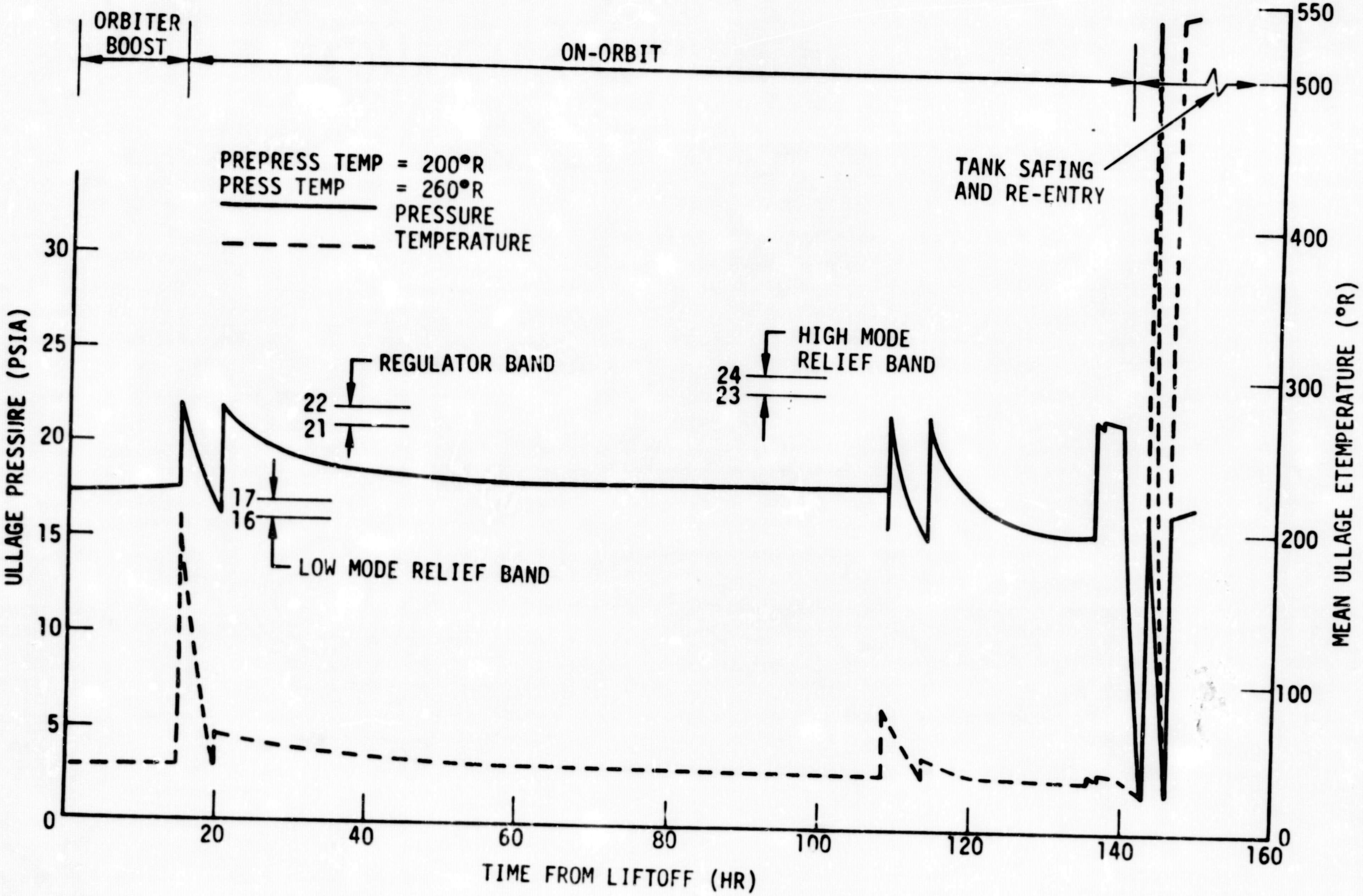


Figure B-6. Main LH_2 Tank Temperature/Pressure History

Purge Bag

The Purge Bag will be designed to accommodate propellant leakage normally associated with joints, seals, and connections. The Purge Bag will not be designed to accommodate failure of propellant tank or propellant tank components (i.e., seals, connections, joints). Leak detection is not required with the Purge Bag operation. Since the Purge Bag is utilized only in ground operations, it will not be designed to the fail safe criteria.

Payload Structural Support Provisions

Primary structural support of the payload while in the Space Shuttle will be from the Space Tug/payload structural interface located at the forward end of the Tug. See Figure B-2. For long payloads, secondary structural supports from the payload to the Space Shuttle cargo bay support points may be incorporated to reduce excessive structural deflections and loads. Secondary structural supports should be designed to preclude the introduction of Shuttle flight loads into the payload and Space Tug.

Structural Analysis

The structure shall be designed to survive the 20 missions with no scheduled structural refurbishment, and in a manner that does not reduce the probability of the successful completion of any mission. Consideration shall be given to the cumulative deteriorating effect of repeated exposure to the critical conditions, such as temperature, creep, and fatigue.

The structure shall be designed by flight conditions wherever possible. The nonflight conditions and environment shall influence the design to the minimum extent possible.

The contractor shall show by analyses that the preliminary structural design meets the design requirements with sufficient margin of safety to assure adequate strength, rigidity, and safety of personnel at all times. The Stage shall be designed to minimize weight and yet resist all loads and combination of loads that may reasonably be expected to occur during all phases of fabrication, testing, transportation, erection, checkout, launch, flight, and recovery. Methods of analysis, material allowables, and formulas shall be adequately referenced to MIL-HDBK-5 or other supplemental documents as may be determined by MSFC to be acceptable standard references. When used in addition to MIL-HDBK-5, supplemental documents shall be in a published form and available to MSFC upon request. The computer programs utilized must be documented and approved by MSFC. The material strength minimum guaranteed values that show no pronounced yield point, the yield point shall be the 0.2 percent offset value. Material strength allowables shall include all environmental effects to which the material will be exposed from fabrication through flight. For materials where the yield point cannot be established, the safety factor against ultimate shall govern.

Definitions

The following definitions and terms shall be used for design and analysis of the stage or vehicle and in all documentation to establish uniform nomenclature with respect to loads, safety factors, etc. The factors of safety to be used in the design and analysis of the structural and propulsion syatems are also listed in this standard.

Limit Load - Limit load is the maximum load calculated to be experienced by the structure under the specified conditions of operation and includes steady state load, using the appropriate acceleration, and dynamic load due to environment or forcing function.

Design Load - The design load is the limit load multiplied by the required minimum factor of safety.

Allowable Load - (Stress) - The allowable load or stress is the maximum load or stress a particular element can be subjected to, as for example: if buckling is the design criterion, then the allowable load on a column is the load which causes buckling.

Factor of Safety - The factor of safety is defined as the ratio of the allowable load or stress to the limit load or stress.

Margin of Safety - The margin of safety is the percentage of which the allowable load or stress exceeds the design load or stress. For example, allowable stress can mean the material yield stress, the material ultimate stress, etc.

Operating Pressure - The operating pressure is the nominal pressure to which the components are subjected under steady state conditions in service operations.

Limit Pressure - The limit pressure is the maximum operating pressure or the operating pressure including the effect of system environment such as vehicle acceleration. For hydraulic and pneumatic components and assemblies, limit pressure will include the effect of pressure transients.

Proof Pressure - The pressure to which every production pressure containing component is subjected. Unless otherwise approved by the procuring agency, the proof pressure shall be run at operating temperatures. Every point in the test article shall be subjected to at least the proof factors specified in Factor of Safety section below. Where practical the proof pressures shall also be used to envelope the combined effect of external load and internal pressure. The component shall not exhibit any deformation detrimental to operation of the component at the proof pressure level.

Yield Pressure - A test level as specified in Factor of Safety section below to which the qualification article of certain pressure containing components is subjected. Unless otherwise approved by the procuring agency the yield pressure test shall be run at operating temperature. The component shall not exhibit permanent deformations other than restrained local yielding pressure.

Ultimate Pressure - A test level as specified in Factor of Safety section below to which at least one qualification article of every pressure containing component is subjected. Unless otherwise approved by the procuring agency the ultimate pressure test shall be run at operating temperature. The component shall not rupture or leak at the ultimate pressure level.

Combined Stresses - Combined stresses are stresses resulting from the simultaneous action of all loads and environments.

Factor of Safety

The following factors of safety are the minimum to be applied. These factors shall be applied to the combined stresses with the following exception:

In circumstances where certain loads have a relieving, stabilizing, or otherwise beneficial effect on structural load capability, the minimum expected value of such loads shall be used and shall not be multiplied by the factor of safety in calculating the design yield or ultimate load. For example, the ultimate compressive load in pressurized vehicle tankage shall be calculated as follows:

Ultimate Load = Safety Factor x Body Loads - Minimum Expected Pressure Load

For components or systems subjected to several missions, safety factor requirements shall apply to the final mission.

(1) General Safety Factors

Manned Vehicle (Tug/Space Shuttle)

Yield Factor of Safety	= 1.10
Ultimate Factor of Safety	= 1.40

Unmanned Vehicle (Tug/Payload)

Yield Factor of Safety	= 1.10
Ultimate Factor of Safety	= 1.25

(2) Propellant Tanks

Manned Vehicle	
Proof Pressure	= 1.05 x limit pressure
Yield Pressure	= 1.10
Ultimate Pressure	= 1.40
Unmanned Vehicle	
Proof Pressure	= 1.05 x limit pressure
Yield Pressure	= 1.10
Ultimate Pressure	= 1.25

(3) Hydraulic or pneumatic systems
Flexible hose, tubing and fittings less than 1.5 inch in diameter

Proof Pressure	2.00 x limit pressure
Burst Pressure	4.00 x limit pressure

Flexible hose, tubing and fittings (including LOX and LH_2 vent lines) 1.5 inch in diameter and greater

Proof Pressure	1.50 x limit pressure
Burst Pressure	2.50 x limit pressure

Gas reservoirs

Proof Pressure	1.50 x limit pressure
Yield Pressure	1.10 x present pressure
Burst Pressure	2.00 x limit pressure

(4) Actuating cylinders, valves, filters, switches

Proof Pressure	1.50 x limit pressure
Burst Pressure	2.50 x limit pressure

(5) LH_2 and LOX Feed Lines

	Shuttle Operation	Tug Operation
Proof Factor of Safety	1.05	1.05
Yield Factor of Safety	1.10	1.05
Burst Factor of Safety	1.40	1.25

Note: The Factors of Safety of (3), (4), and (5) are never used in combination with those shown under (1) and (2).

When a pressurized system or component is subjected to external loads, such as air loads, ground handling, transportation, in addition to pressure, factors of safety given above will be used. That is, the pressure vessel thickness is determined by the use of applicable pressure factors and then the component is analyzed for the external loads pressures, and environments with the general safety factor.

When adequate fracture toughness data and sufficient knowledge of operating conditions are available to determine the required proof pressure from fracture mechanics principles, the required proof pressure may be determined from this data and used instead of the safety factors listed above. Written approval by MSFC will be required.

Fatigue Analysis

All structural elements shall be evaluated for their capability to sustain any cyclic load condition which is part of the design environment For those elements whose design is controlled by a cyclic or repeated load condition, or a randomly varying load condition, a preliminary fatigue analysis will be conducted if sufficient knowledge of the operating conditions are available.

If sufficient fatigue data are available to establish statistical minimum guaranteed fatigue allowables, the component shall be capable of withstanding three times the predicted number of load cycles. If only typical fatigue allowables are available the component shall be capable of withstanding ten times the predicted number of load cycles. For cyclic loads to varying levels such standard methods as Miner's method shall be used to determine the combined damage. For repeated load combined with a steady load such standard methods as the modified Goodman diagram shall be used to determine the combined effect.

AVIONICS

The Tug avionics system as defined in Figure B-7 and Table B-9 will be designed for minimum weight and to satisfy the following limitations:

a. Provisions will be made for monitoring of Tug critical functions, for Shuttle crew safety, to be monitored by the Shuttle crew at all times the Tug is attached to the Shuttle.

b. Tug avionics will be in an unpowered condition during Shuttle ascent and descent operations except for safety and thermal requirements.

c. The Tug will be provided a navigation update from the Shuttle prior to Shuttle/Tug separation.

d. Communication system will be designed to be compatible with MSFN and Shuttle Deep Space Network.

e. No avionics interfaces between Tug and payload.

f. Tug avionics will be designed for a 20 mission life with minimum refurbishment after each mission as required.

g. Tug will be designed for autonomous operation except in areas where remote control will significantly reduce weight without compromising operational effectiveness.

h. Autonomous guidance, navigation, and control (with ground or Shuttle orbiter override capability).

   - For low earth orbit within 10 KM, 5 M/Sec
   - For synchronous orbit within 50 KM, 5 M/Sec

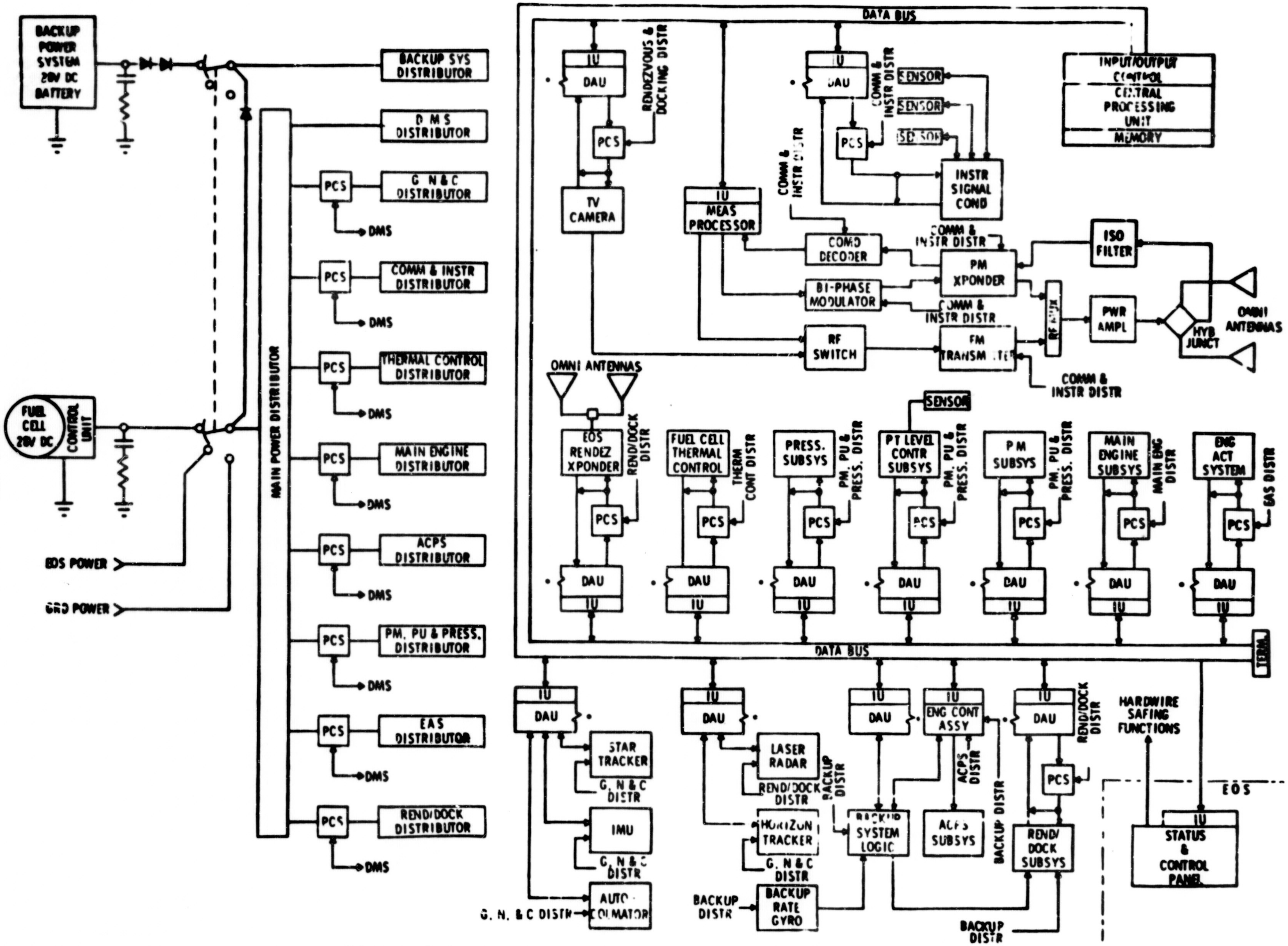


Figure B-7 Avionics Subsystem Integrated Block Diagram

Table B-9. Sync-Orbit Tug Avionics Equipment

GN&C	FUNCTION	SPECIFICATIONS
3 Axis Rate Gyro Pkg	Stab. & Course Alignment	0.1°/Hr - 0.3°/Hr/G
2 Gimballed Star Tracker	Inertial Coordinate Frame	30 Arc Sec
Horizon Tracker	Radius Vector Direction & Mgtd	0.1° low orbit
Accelerometers	Engine Thrust Vector Control	0.1 M/Sec Resolution
Laser Radar	Rendezvous	0.1 M Range, 0.1°
Television	Docking	250 lines, 15 frames/Sec.
FM Transmitter	Docking	500 KHz
COMM. & DATA MGMT		
Sensors (300)	Data Acquisition	- - -
Digital Interface Units	Signal Conversion	32 Channel
Telemetry Formatter	Data Interfacing	200 K Bits/Sec
Computer	Digital Processing	16 K Memory (Expandable)
Command Decoder	Remote Control	200 - 1000 Commands/Sec
S-Band Transponder	Ranging Tele. & Command	52 K Bit Tele. & 1 K Bit Data
Mod/Demod Processor	Process Transmitter Info.	- -
ELECTRICAL POWER		
Fuel Cells	Primary Electrical Power	28 V. DC ± 5%, 1 KW
Battery	Emergency Power Attitude Hold & Etc.	28 V. DC Pwr for 30 Min.
Control and Distribution	Control & Distribution of Elect. Pwr	- -
O_2 and O_2 Tank	Cryo Storage for Fuel Cell	O_2 99.99%, 1 PPM CO & CO_2
H_2 and H_2 Tank	Cryo Storage for Fuel Cell	H_2 99.98%, 1 PPM CO & CO_2

i. Rendezvous laser required with capability of detecting and tracking a satellite corner reflector from 100 KM within 0.1 meters and 0.1 degrees.

j. Rendezvous to 300 meters automatic; closure and docking with man-in-the Loop TV.

k. Attitude maintained during burns to within 0.1 degrees of the desired attitude.

l. Sustaining power to Tug during ascent and descent is Shuttle-furnished (up to 300 W avg., 500 W peak).

m. Digital multiplexing techniques will be utilized to interconnect Tug subsystems with the data management subsystem.

PROPULSION

The Tug propulsion system will be designed to satisfy the following limitations:

a. Main engine, see Table B-10.

Table B-10. Space Tug Main Engine Ground Rules

Number Main Engines	1
Propellants	Liquid Oxygen/ Liquid Hydrogen
Minimum Vacuum Thrust, Pounds	10,000
Nominal Engine Mixture Ratio	6.0:1
Engine Mixture Ratio Range	5.5 - 6.5
Engine Chamber Pressure, PSIA	2020
Engine Length/Diameter, Inches	68/36
Vacuum Thrust Throttling Capability	5.0:1
Nozzle Configuration	Bell
Nozzle Expansion Ratio	400:1
Turbine Drive Cycle	Staged Combustion
Min. Guaranteed Vacuum Specific Impulse, Seconds	470
Nominal Specific Impulse, Seconds	473.8
Engine System Weight, Pounds	213
Number of Vacuum Starts	160
Service Life Between Overhauls (Reusable Mode), Thermal Cycles	300
Service Life Between Overhauls (Reusable Mode), Hours	20
Gimbal Angle (Square Pattern), Degrees	±7
Gimbal Acceleration, Radians/(Seconds)^2	20
Minimum Natural Frequency of Gimbal System, Hertz	10
Fuel Pump NPSH, Feet of Hydrogen	15
Oxidizer Pump NPSH, Feet of Oxygen	2
Maximum Single Run Duration, Seconds	1400
Maximum Storage Time in Orbit (Dry), Weeks	1
Maximum Time Between Firings (Coast Time), Days	3
Minimum Time Between Firings (Coast Time), Minutes	10
Service-Free Engine Run Time, Hours	1
Service-Free Engine Firing Cycles	10

b. The APS system will be designed for minimum weight to meet the accuracy and impulse requirements shown in Avionics i, k, and Table B-11 and Table B-3, respectively.

c. There is no direct physical interface between the Tug propulsion system and the payload.

d. While in orbit after separation from the Shuttle, the Tug will provide attitude control for the Tug - Payload assembly until Tug - Payload Separation. The Tug will not provide attitude control for experiment pointing.

e. There will be no propellant sharing between Tug and payload.

f. The baseline synchronous equatorial mission will determine propulsion system design and operational characteristics.

g. The Tug shall be capable of being loaded with propellants, pressurants and other fluid reactants while in the Shuttle cargo bay on the launch pad. This will be accomplished through fill and drain systems that are separate from those of the Shuttle, but accessible with Shuttle on the pad in the vertical position (Tug inverted) with the cargo bay doors closed.

h. Propellant loading shall be accomplished in such a manner that no contaminants are introduced into the Shuttle cargo bay.

i. The Tug shall be capable of safely venting propellant boiloff gases while on the launch pad, during launch and flight, in orbit, and during reentry while still in the Shuttle cargo bay.

k. In the event of abort to orbit, the Tug shall have the capability of safely dumping propellants prior to Shuttle orbiter landing. Propellant dump provisions should be provided only during the orbital coast phase of an abort to orbit mode. Acceleration for Tug propellant settling is provided by Shuttle.

l. The propulsion system of the Tug will be remotely checked out while in the cargo bay both prior to launch and while in orbit prior to separation from the Shuttle.

m. There will be no propellant sharing between Tug and Shuttle.

n. Tug propulsion system prestart functions will be accomplished subsequent to deployment but prior to Tug and payload/Shuttle separation.

o. The ACPS and main engine (idle mode) can be fired after deployment from the orbiter but while still attached to the orbiter.

p. All Tug propulsion systems will be required to be in a safe condition prior to reentry from orbit in the Shuttle.

q. After retrieval from orbit, the Tug propellant vent and purge interface must be re-established.

r. Prior to, or during reentry, the Tug propellant tanks must be purged of residuals. Purge gases are to be stored in the Shuttle cargo bay.

s. The main engine will be used for orbital injection and braking.

t. RCS system will be used for small rotational and translational maneuvers and will use GO_2/GH_2 as the propellants ($I_{sp} = 380$).

u. The main engine at 20 percent thrust ($I_{sp} = 461$) may be used on-orbit for large (>100 ft/sec) translational maneuvers.

v. Contaminants from Tug RCS thrusters shall not impinge harmfully on payload.

w. Main engine pressurization system and RCS system will be designed as an integrated subsystem.

x. The oxygen and hydrogen required by the fuel cell system will be stowed in common tankage with the RCS.

THERMAL

The Tug thermal control system and cryogenic insulation system will be designed to satisfy the following limitations:

a. Thermal Control System

(1) The Tug thermal control system will be designed to maintain all temperature critical subsystems within their operating temperature range during all phases of flight from lift-off to landing. A design objective is to use a passive thermal control system for the avionics (Except for fuel cell).

(2) The thermal control system of the Tug shall not provide thermal control to payload systems.

(3) The thermal control system of the Tug shall not require selective orientation in orbit to perform the thermal control functions.

(4) Active thermal control to Tug systems is not required from the orbiter during flight operations.

b. Cryogenic Insulation

(1) A fluid umbilical shall be provided in the orbiter cargo bay to supply a dry nitrogen and a helium purge of the cryogenic insulation during ground hold.

Table B-11. Equatorial Synchronous Orbit - ΔV Budget

EVENT	V FT/SEC		
	MAIN ENGINE	ORBIT MANEUVER	RCS
SEPARATE FROM SHUTTLE AT 100 N.MI.			10
PERIGEE BURN	8136		
GRAVITY LOSSES	310		
MID-COURSE CORRECTION			50
APOGEE BURN	5883		
GRAVITY AND TURNING LOSSES	10		
STATION KEEPING			30
DEPLOY PAYLOAD			10
INJECT INTO PHASING ORBIT FOR RETRIEVE PAYLOAD		100	
RETRIEVE PAYLOAD		100	15
DEORBIT	5814		
GRAVITY LOSSES	7		
MID-COURSE CORRECTION			50
CIRCULARIZE IN 270 N. MI.	7842		
GRAVITY LOSSES	25		
TRANSFER TO SHUTTLE ORBIT 100 N.MI.	592		
TERMINAL RENDEZVOUS		100	15
DOCK WITH SHUTTLE AT 100 N.MI.			10
CONTINGENCY (2 PERCENT)	572		
TOTAL	29,191	300	190

(2) The cryogenic insulation of the Tug shall be consistent with a maximum orbiter cargo bay wall temperature (See Table B-5).

(3) The orbiter cargo bay shall provide for overboard venting of Tug propellant boiloff and for insulation purge gas outflow, and for helium repressurization during reentry.

(4) The surface characteristics of the Tug cryogenic insulation shall not be materially degraded to contaminant gases in the orbiter cargo bay.

(5) Cryogenic temperatures of the Tug propellants shall be isolated from the Tug subsystems and Tug payload by cryogenic insulation and by heat blocks in the structure.

c. Thermal Design Environment

The orbital heating environment for the Tug shall be determined to be consistent with the mission time line given in Table B-2.

INTERFACES

TUG/SHUTTLE

Physical Interface

The Shuttle Orbiter cargo bay shall provide a clear volume required to accommodate a payload of 15 feet in diameter and 60 feet in length. The volume requirements for the mechanism and attach points to the Tug in addition to that required for Tug support and restraining mechanism and attach points shall be minimized.

The Tug/Shuttle Orbiter docking mechanism and hardpoints for forward frame and possibly for Shuttle Orbiter remote Manipulators make up the Tug/Shuttle structural interface. The Shuttle Orbiter payload attach points are undefined, however, the Tug will be supported from two Shuttle Orbiter provided hard points located on the payload bay longerons in the area of the Tug interstage adapter. Load distribution from the Tug to these hard points will be accomplished by the Tug interstage adapter. The Shuttle Orbiter will be assumed to provide the Tug deployment system. Secondary supports and required load qualization system necessary to preclude load interaction between the Tug and Shuttle Orbiter will be chargeable to the Shuttle Orbiter payload capability. The Shuttle Orbiter payload deployment mechanism design will not be defined in this study. The Tug will be mounted with the engine in the forward end of the cargo as far forward as possible to maximize Tug payload length capability, see Figure B-3.

The physical interface between the Tug and Shuttle Orbiter will include: Fluid, electrical, mechanical and structural interfaces. Fluid interfaces identified initially are propellant tanks fill/drain/vent/dump, tank inerting pressurization, HPI vent out-flow and repressurization, fuel cell vent, and Tug ground conditioning. Fluid interface connections shall be available in the Shuttle Orbiter cargo bay as shown in Schematic diagram of Figure B-8.

A hardline electrical interface between Tug and Shuttle Orbiter will be utilized to provide power to the Tug during the missions phases of boosting the Tug/payload to 100 NMI orbit and returning the Tug/payload to Earth from the 100 NMI operations orbit. The interface provisions (hardline or RF link) for transmitting navigation update to the Tug and transmission of Tug system status information to the Shuttle are not yet defined. These functions could be accommodated by either or both modes of data transmission.

Functional Interface

The Tug, its payload and any Tug-peculiar auxiliary equipment will be carried to and from the Tug operations orbit (100 NMI, 28.5 inclination) within the Shuttle Orbiter cargo bay. A maximum payload of 65,000 pounds will be

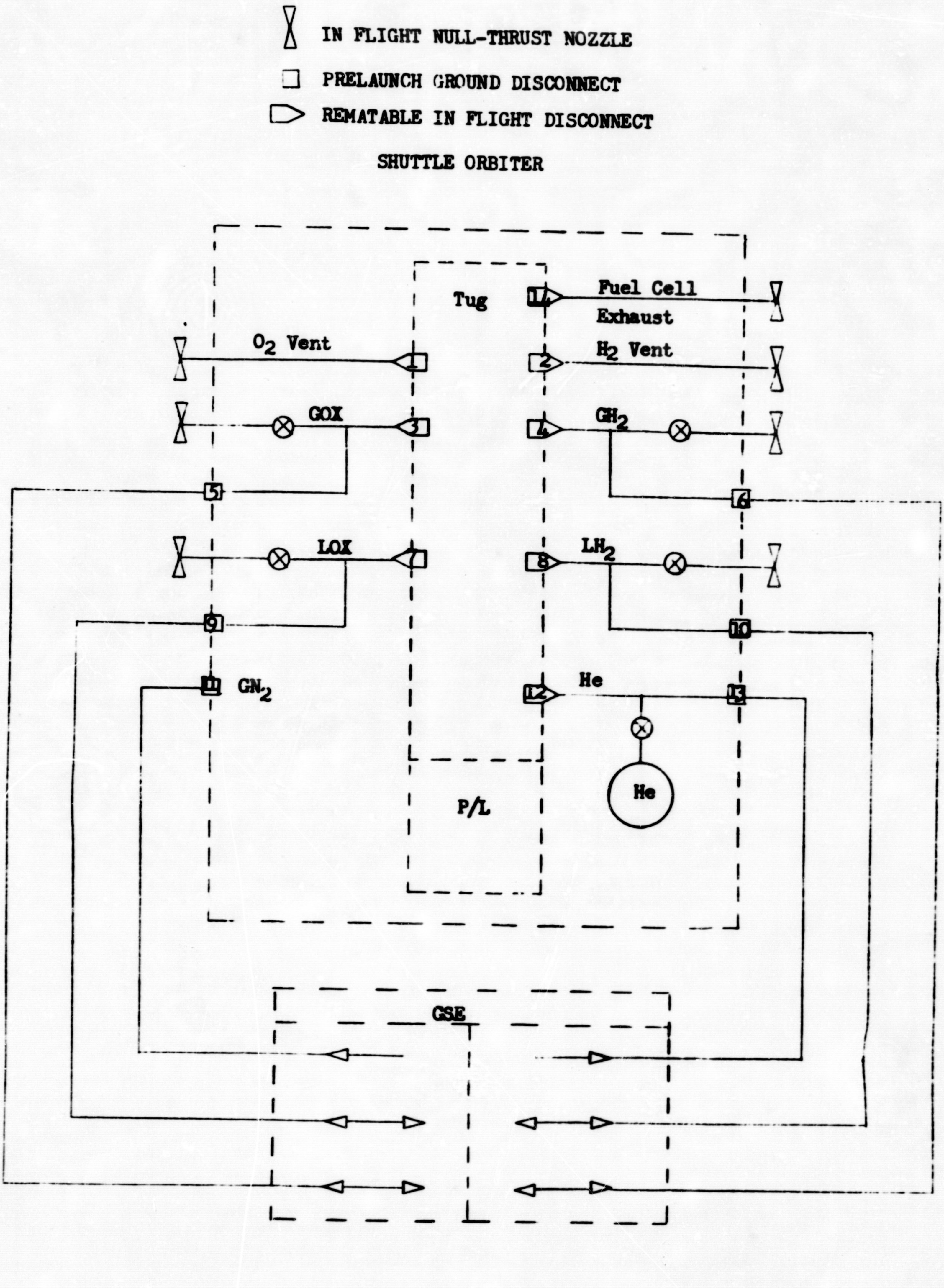


FIGURE B-8. FLUID INTERFACE SCHEMATIC

delivered to the Tug operations orbit by the Shuttle Orbiter with C.G. limits within the Shuttle Orbiter cargo bay as shown in Figure B-9 between liftoff and landing, payload venting shall be non-propulsive and shall not impart disturbances greater than 10 nor total impulse greater than 1000 lb-sec.

Release of the Tug from the standard deployment mechanism shall leave the Tug and Shuttle Orbiter in a stable mode with minimum disturbing torques. The Tug RCS propulsion shall be inhibited while attached to the Shuttle including stowage and retrieval phases.

After physical separation, the Tug/payload shall hold attitude until the Shuttle Orbiter has assumed a predetermined remote position. In the event of a Tug abort following separation from the Shuttle Orbiter, the Tug will provide an indication to the Shuttle Orbiter crew or ground which indicates its safety condition relative to a recovery by the Shuttle Orbiter on another Tug.

The Tug/payload shall perform a passively cooperative role during Shuttle Orbiter rendezvous and dock as specified in "Flight Criteria" items g and h.

Environment to which the Tug will be exposed while in the Shuttle Orbiter cargo bay is defined the "induced environment" section of the design criteria.

During ground operations, the Tug and payload will be installed or removed from the Shuttle Orbiter as an integral unit, i.e., in the mated configuration. This applies to installation/removal operations with the Shuttle Orbiter in either the horizontal attitude prior to mating with the Shuttle Booster or in the vertical attitude following Booster/Orbiter mating operations.

TUG/PAYLOAD

Physical Interface

The Tug/payload docking mechanism comprises the physical interface between the Tug and payload. The weight associated with payload racks, interstage structure adapters, separation mechanism, and any other payload-related equipment including the passive docking provisions shall be chargeable to the payload. The payload center of gravity is defined as being at the geometric center of the 15 by 25 post payload envelope. There shall be no fluid or electrical connections to the payload.

Functional Interface

The Tug shall transport an autonomous payload from the 100 NMI, 28.5° inclined orbit to a predesignated position in geosynchronous orbit and retrieve a payload in geosynchronous orbit and return it to the 100 NMI, 28.5° inclined Tug operations orbit. Separation from the payload shall be accomplished at the destination with minimum disturbances imparted to the payload. The Tug will be designed to rendezvous and dock with a payload that has been designed for recovery. The payload is assumed as a passive but 3-axis stabilized vehicle with a docking mechanism compatible with the Tug. Tug rendezvous with the payload will be automatic to within 300 meters with man-in-the-loop for final closure and docking.

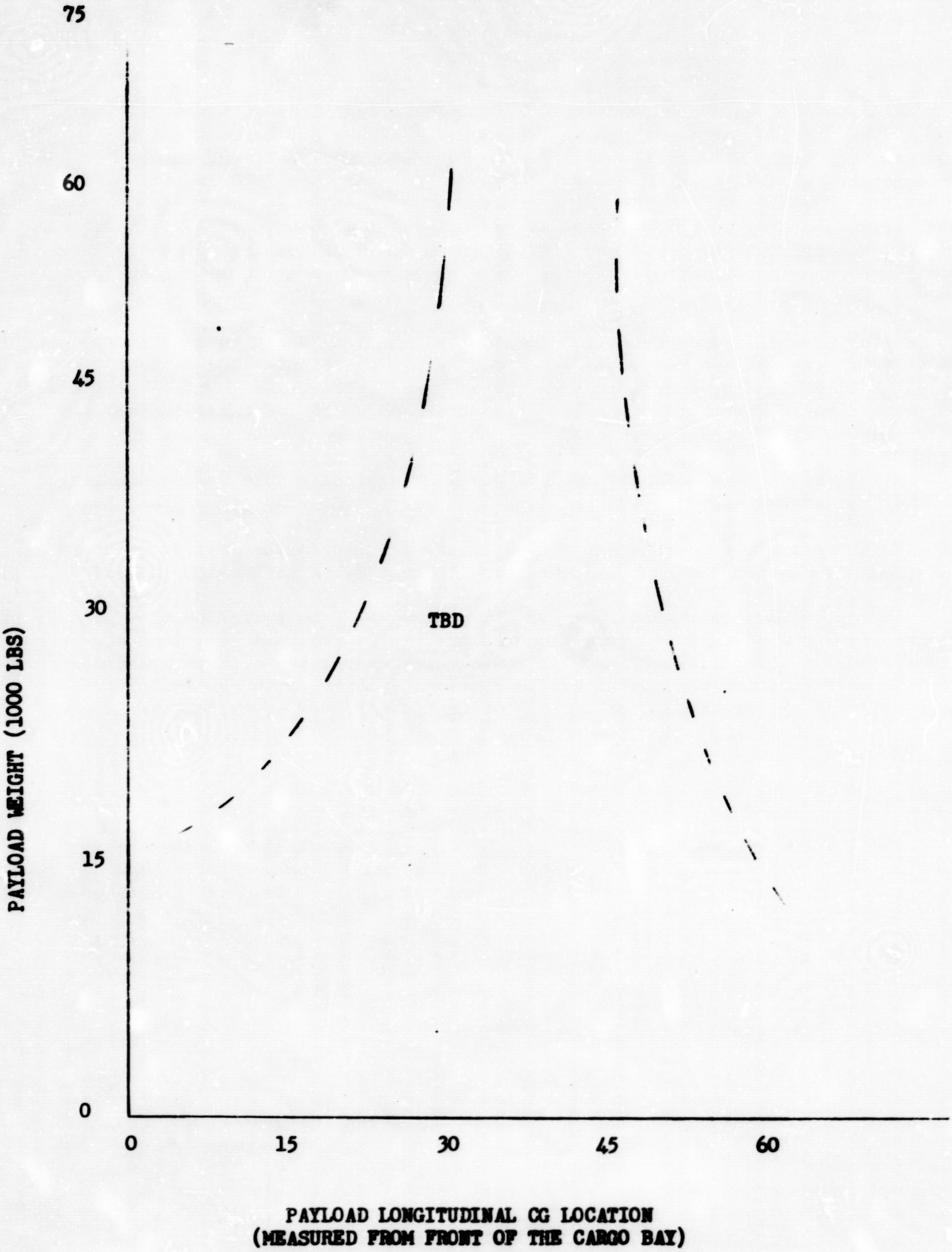


FIGURE B-9. ORBITER PAYLOAD C.G. ENVELOPE

The Tug is required to interface with other payloads in alternate missions for example, the Tug is required to transport one way an 8.06K payload to geosynchronous orbit and return without payload for rendezvous with the Shuttle Orbiter. For this mission, a sample payload separation interface only is required rather than a more complex and heavier docking interface. Another mission would require the Tug to proceed without payload to geosynchronous orbit where it would rendezvous and dock with a 4.16K payload to be returned to the Shuttle Orbiter.

TUG/GSE

Physical Interface

A physical interface between Tug and GSE will exist during in-process and post-fabrication checkout, test, transportation, and eventual Shuttle-Tug loading/unloading operations. Hard points shall be provided on the Tug structure to facilitate post-fabrication handling and transportation. These hard points should be the same as those utilized in Shuttle Orbiter cargo bay support and deployment operations, if possible. Transportation and handling loads will not exceed flight loads. However, if the Tug is dependent upon tank pressurization for its rigidity, provisions must be made on the Tug structure for mounting "strong back" GSE to support the Tug when it is depressurized. Provisions must be made for Tug checkout to verify operational integrity of the Tug systems and subsystems following fabrication test, transportation, storage and maintenance. Provisions will be made in the subsystems designed to facilitate checkout and servicing functions through supplying adequate test points, service ports and/or umbilicals.

Functional Interface

The ground support equipment will interface with the Tug to provide Tug checkout, servicing, handling and transportation functions. In order to minimize interface requirements between Tug and GSE, maximum utilization will be made of the on-board checkout capability built into the Tug.

TUG/GROUND FACILITIES

The Tug will have physical and/or functional interface with ground facilities during fabrication, test, mission operations, and maintenance/refurbishment. The facilities with major design impact on the Tug are the mission support facilities (i.e., MSFC, Deep Space Network). The Tug avionic system data format, data rate, and transmission power must be compatible with the mission support facilities.

GROUND SUPPORT EQUIPMENT CRITERIA

Ground Support Equipment becomes an extension of the Tug systems during checkout, maintenance, servicing, handling, and transportation of the Tug. The following GSE related design criteria should be implemented in Tug design whenever possible without appreciable weight penalty to Tug:

a. Locate connections to be accessible from the Ground or from simple access equipment. Consider the possibility of impact damage to the Tug vehicles by servicing equipment, and minimize such exposure through optimum placement of service locations.

b. Provide retention or latch devices for umbilicals that might inadvertently disconnect (because of their weight).

c. Avoid use of loose access plates, hardware and filler caps. Use captive devices (chains, cable, etc.).

d. Locate hinges so that access doors will swing clear, preferably down, of swinging point; and so that opened or unlatched doors will be visible and obvious.

e. Locate the service connections to avoid congestion, and to permit simultaneous servicing of all functions (and avoid the need for sequencing ground service functions).

f. Provide specific location(s) and fitting for the attachment of a grounding cable to the vehicle. If necessary, provide additional fittings to allow grounding when the vehicle is in any of its operational orientations. If individual servicing operations, require grounding for bonding to the vehicle, provide specific features for this purpose. These special features are required to insure a good electrical connection and to avoid connection of alligator-type grounding clamps at points where the vehicle structure could be scratched or damaged.

g. Minimize the number of and simplify installation requirements for servicing umbilicals to reduce work load during servicing preparations, increase fluid system leak integrity and decrease personnel exposure.

h. Make connections unique to particular fluids to preclude inadvertent or improper use of fluids, and use existing standards for fluid connections when possible.

i. Design connections to avoid contamination from dust, dirt, and moisture, where practical; use pressure fill connections rather than gravity fill which requires opening of the system.

j. Design connections to avoid undesirable bending loads on fittings due to weight of umbilicals and hoses.

k. Because of the expected life period for the Tug and possible environmental conditions in ground operations, special attention to protection against fungi growth and attack is recommended. Consider incorporating a fungicide in susceptible materials, using a volatile fungicide, using non-nutrient materials, or using coatings that are moisture and fungus proof.

FACILITIES

ENVIRONMENTAL CONTROL

Aerospace environmental facilities will be provided for fabrication, assembly and checkout of Space Tug. Rated "clean room" environments will not be required except for special processes and certain component buildup operations covered by discrete process specification callout.

Level III maintenance/refurbishment operations (as defined in Space Tug M&R plan) will normally be performed in the same facility where original fabrication/assembly of item was performed. Environmental control will not exceed that of the original operation.

DEVELOPMENT AND MANUFACTURING

Development Technology Capability

To be Determined.

Fabrication Processes

To be Determined.

OPERATIONS SUPPORT

Level I and Level II maintenance/refurbishment operations will be performed at the same facility site as Tug checkout operations performed prior to Tug/payload mating.

Aerospace environmental facilities will be provided for checkout, mating and other prelaunch-pad operations and for maintenance and refurbishment. Rated clean room environments will be provided only for those checkout and maintenance operations covered by process specification environmental control requirements.

GSE used for transport/handling of Tug prior to payload mating will be suitable for use during Level I maintenance operations.

GSE provisions will be required to monitor pressurization of main propellant tanks during transport and prelaunch-pad operations.

Post delivery operations prior to Shuttle cargo bay insertion will normally be performed in a horizontal mode.

Space Tug will be inserted in Shuttle cargo bay with payload mated. Normal cycle of Tug/payload insertion into cargo bay will be with Orbiter in horizontal mode. Any required removal during launch pad cycle will be in Tug/payload mated configuration. Tug will be de-fueled before removal.

Ground Operations

Servicing and Checkout	TBD
Safing, Handling and Storage	TBD
Launch Operations	TBD
Maintenance/Refurbishment	TBD

Flight Operations Support

Communications (MSFN, Shuttle DSN)

Data Handling and Processing — TBD

MANUFACTURING

To Be Determined

VOL II

APPENDIX C

INTERFACE REQUIREMENTS

Space Division
North American Rockwell

SHUTTLE/TUG INTERFACE REQUIREMENTS

INTERFACE CONTROL DOCUMENT

Space Division
North American Rockwell

Shuttle/Tug Interface Requirements

Interface Control Document

Contents

1.0 SCOPE

This document specifies the interface requirements between the Space Shuttle and the Tug. The interface areas specified herein include structural, electrical, fluid and mechanical.

2.0 APPLICABLE DOCUMENTS

The following documents form a part of this interface control document.

	System Specification for the Space Shuttle
	System Specification for the Space Tug
*Similar to V7-941514	Interface Control Drawing, Shuttle/Tug

3.0 ABBREVIATIONS AND ACRONYMS

HPI	High Performance Insulation
GSE	Ground Support Equipment
n.m.	nautical mile
g	gravitational force

4.0 FUNCTIONAL REQUIREMENTS

4.1 Fluid Requirements

Fluid interfaces identified at this time are propellant fill/drain/vent/dump, tank inerting pressurization, HPI vent out-flow and repressurization, fuel cell vent, and Tug ground conditioning.

4.1.1 Fluid Leakage

No Shuttle cargo shall be permitted to leak, vent, or discharge propellants into the cargo bay.

4.1.2 Propellant Dump

All residual propellant from the Tug shall be dumped while in the Shuttle cargo bay.

4.1.3 Fluid Interface Connections

Fluid interface connections as shown in Figure*(Similar to V7-941514) shall be provided and made available in the Shuttle cargo bay.

*Figures 2.11-1, 2.11-3 & 2.11-4 of Volume III.

4.2 Structural Requirements

4.2.1 Loads

The Tug will be supported by the Shuttle with hard points located on the Shuttle cargo bay longerons and in the area of the Tug interstage adapter. The maximum interface structural loads shall be as defined in Table C-1.

4.2.2 Tug Deployment

The Tug deployment mechanism shall be provided by the Shuttle. Orientation of the Tug to the Shuttle cargo bay shall be as defined in Figure C-1.

4.2.3 Secondary Supports

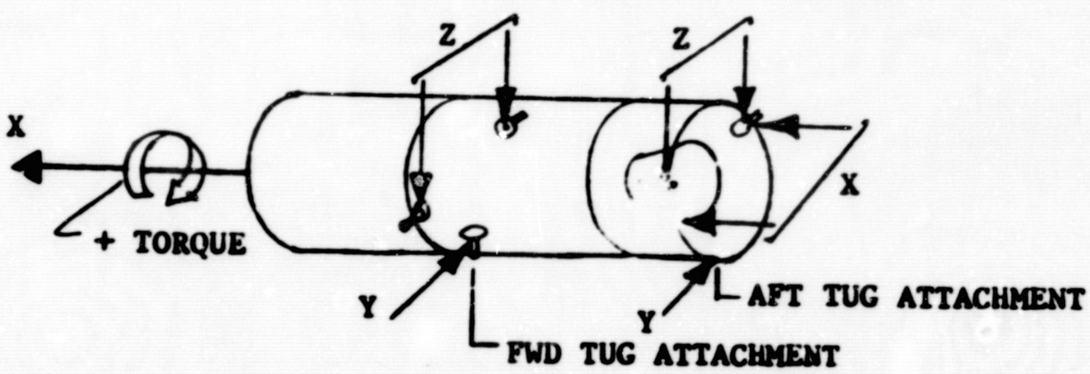
Secondary supports and required load equalization system necessary to preclude load interaction between the Tug and the Shuttle shall be chargeable to the Shuttle payload (Tug) capability (see Figure C-1).

4.2.4 Tug Mounts

The Tug will be mounted on the Shuttle with the engine in the forward end of the cargo bay as far forward as possible to maximize Tug payload length capability.

Table C-1. Space Tug Attachment Limit Loads

Forward Tug (Aft Orbiter)	
Y - Axis:	±35 kips
Z - Axis:	±35 kips
Torque:	±3.20 x 10^6 In.-Lb
Aft Tug (Forward Orbiter)	
X - Axis:	-205 kips
Y - Axis:	±36 kips
Z - Axis:	±36 kips



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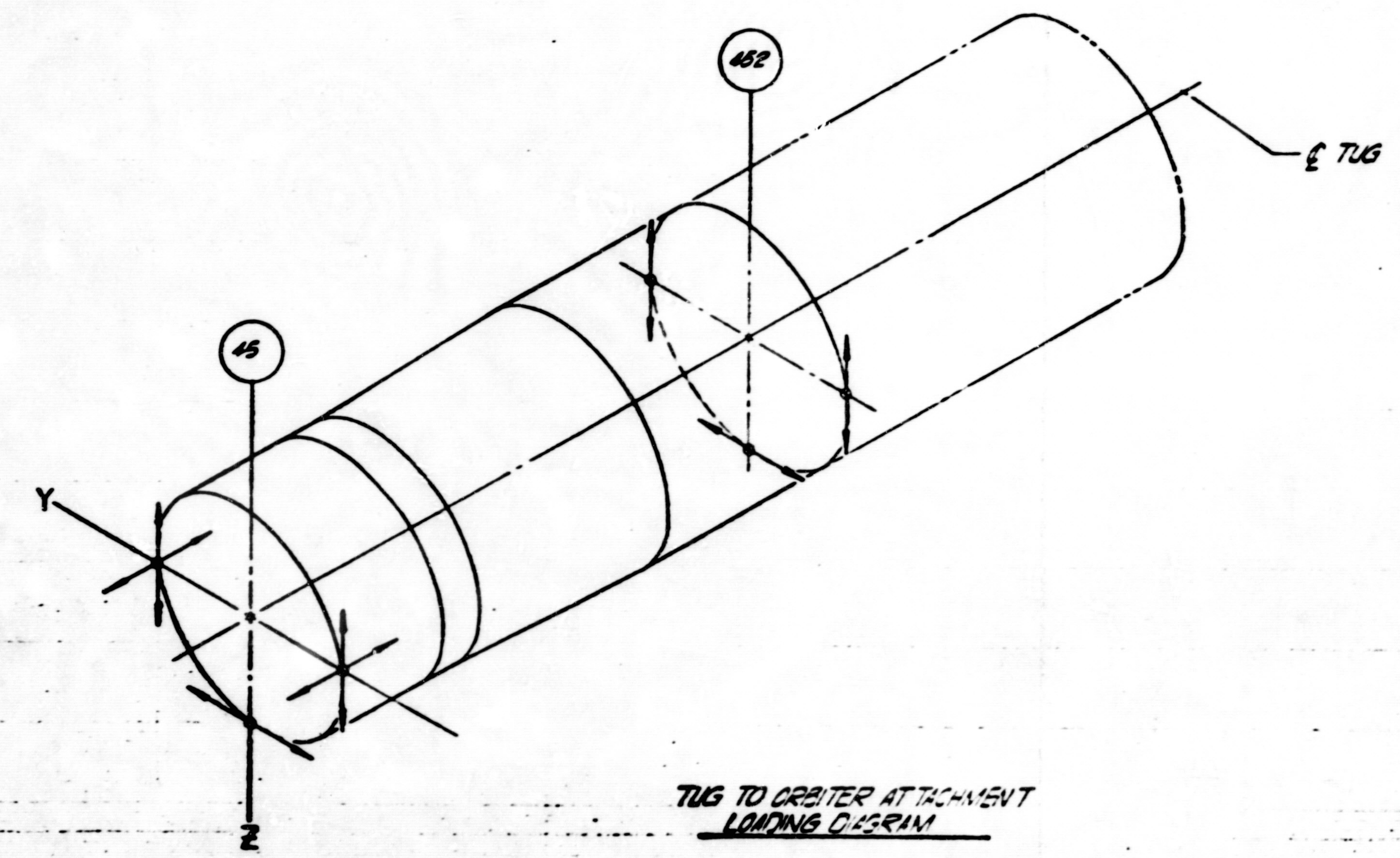
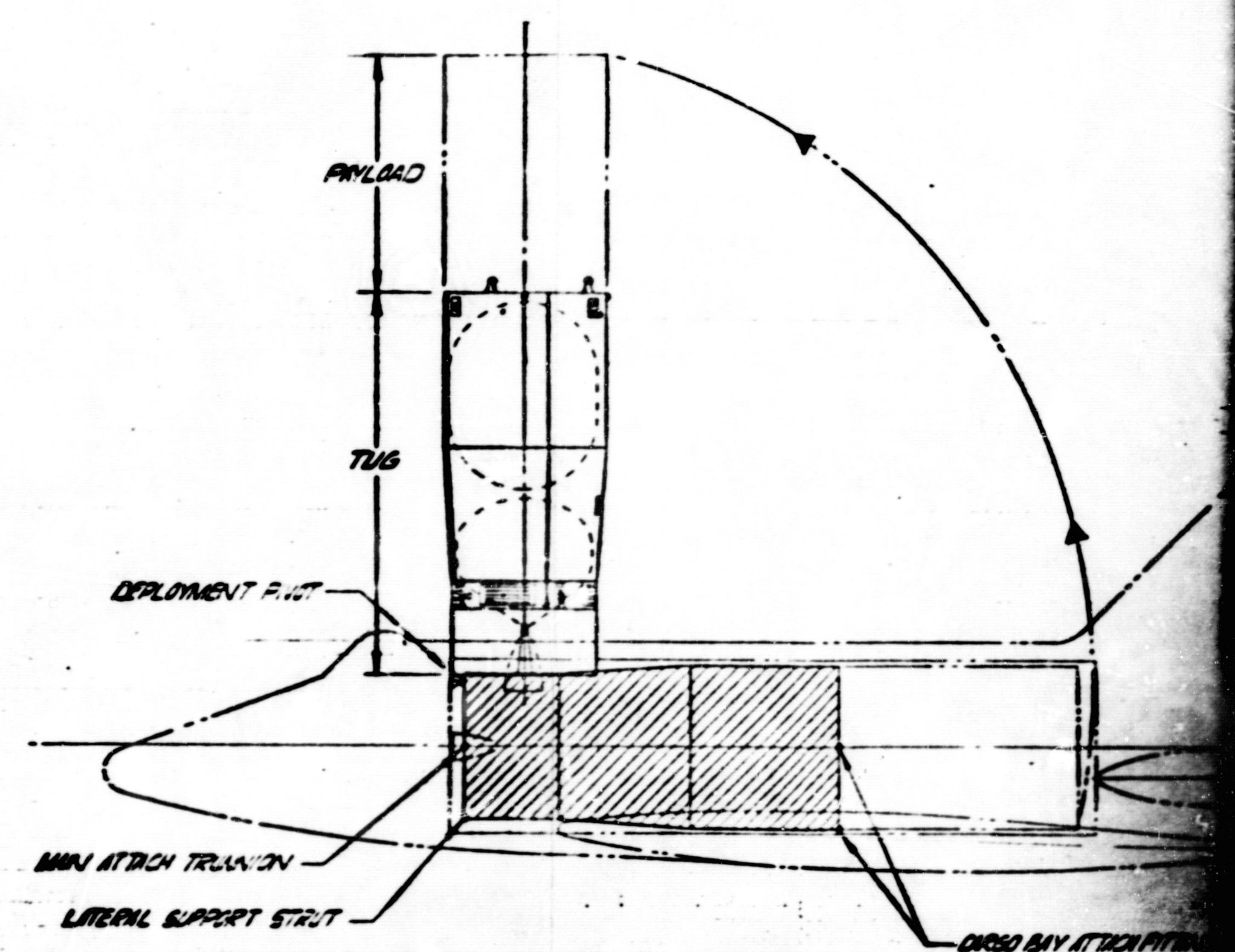
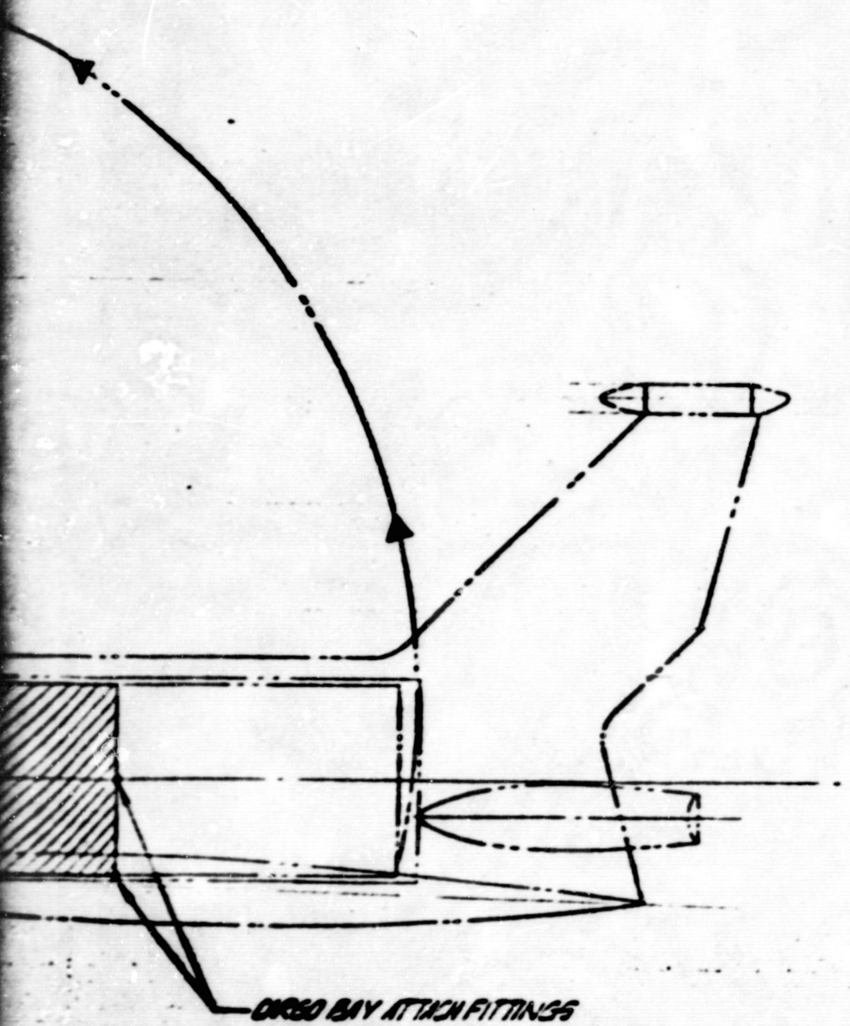


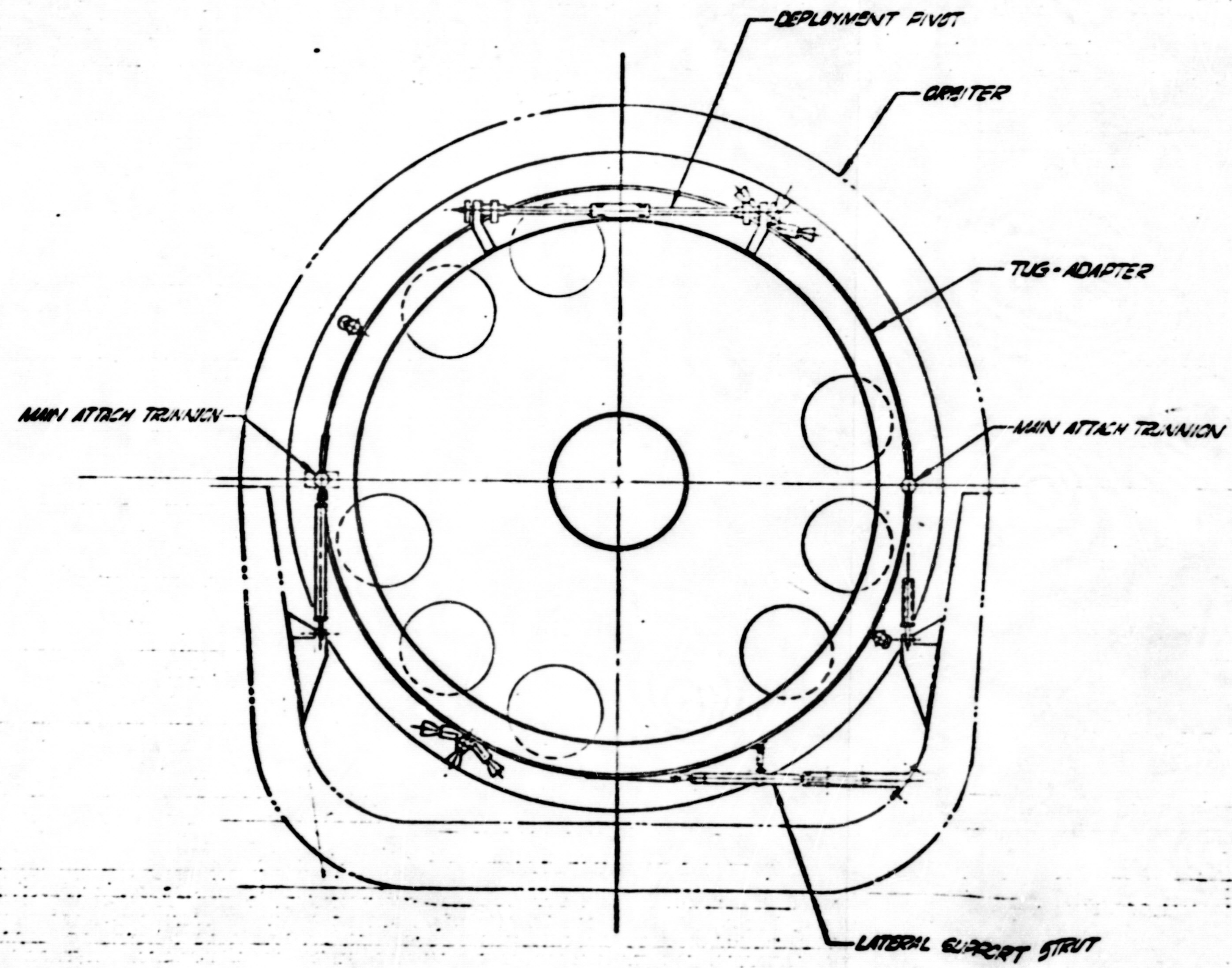
TUG TO ORBITER ATTACHMENT LOADING DIAGRAM



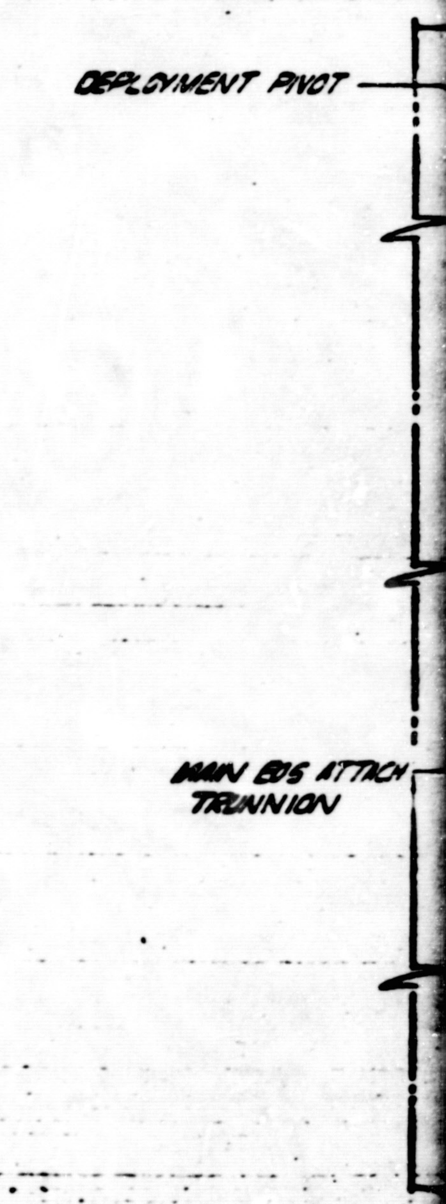
TUG/ORBITER ORIENTATION
SCALE 1/100

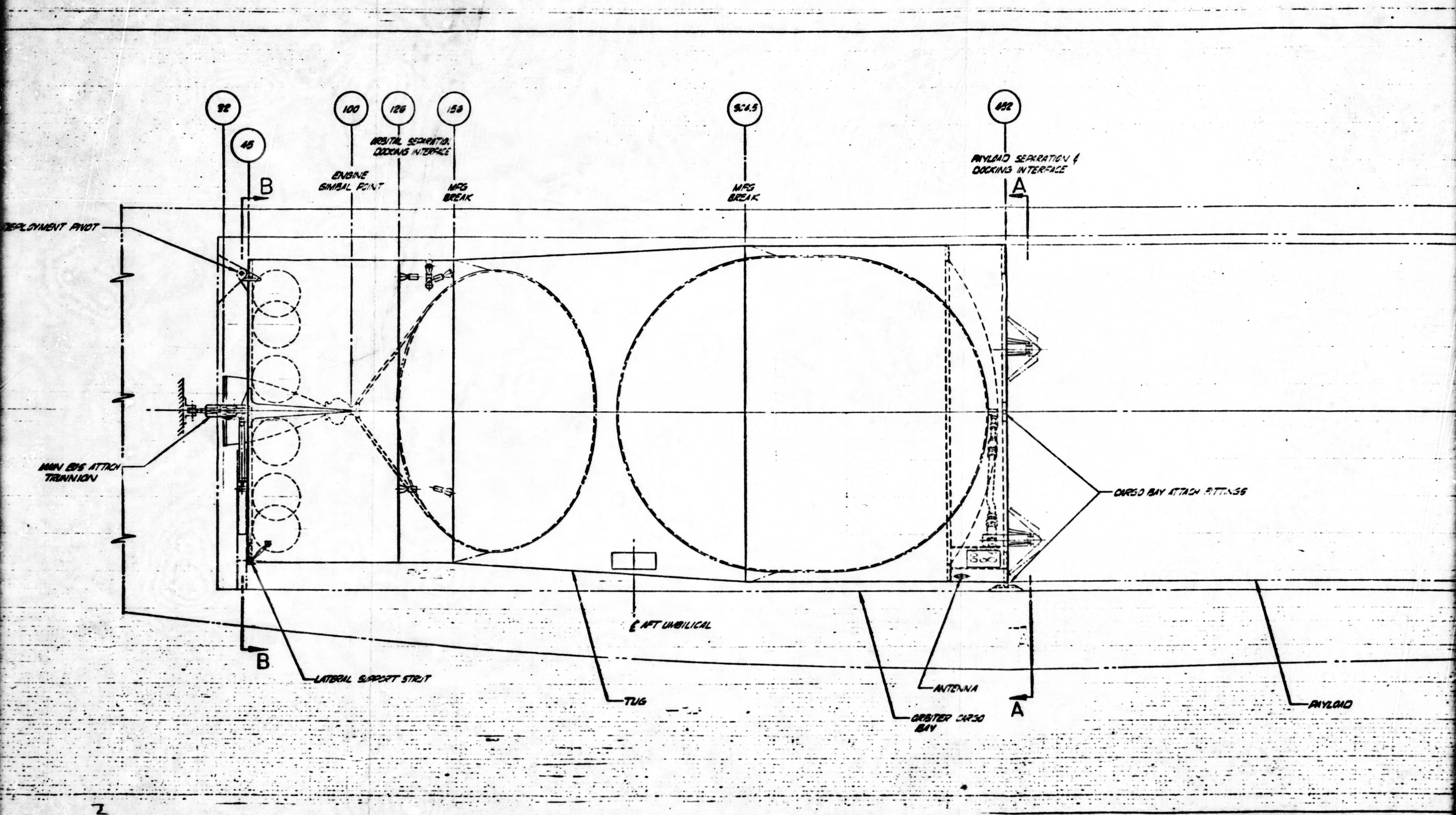


TER ORIENTATION
1/100



SECTION B-B
SCALE 1/20





2

POOR FILMING QUALITY

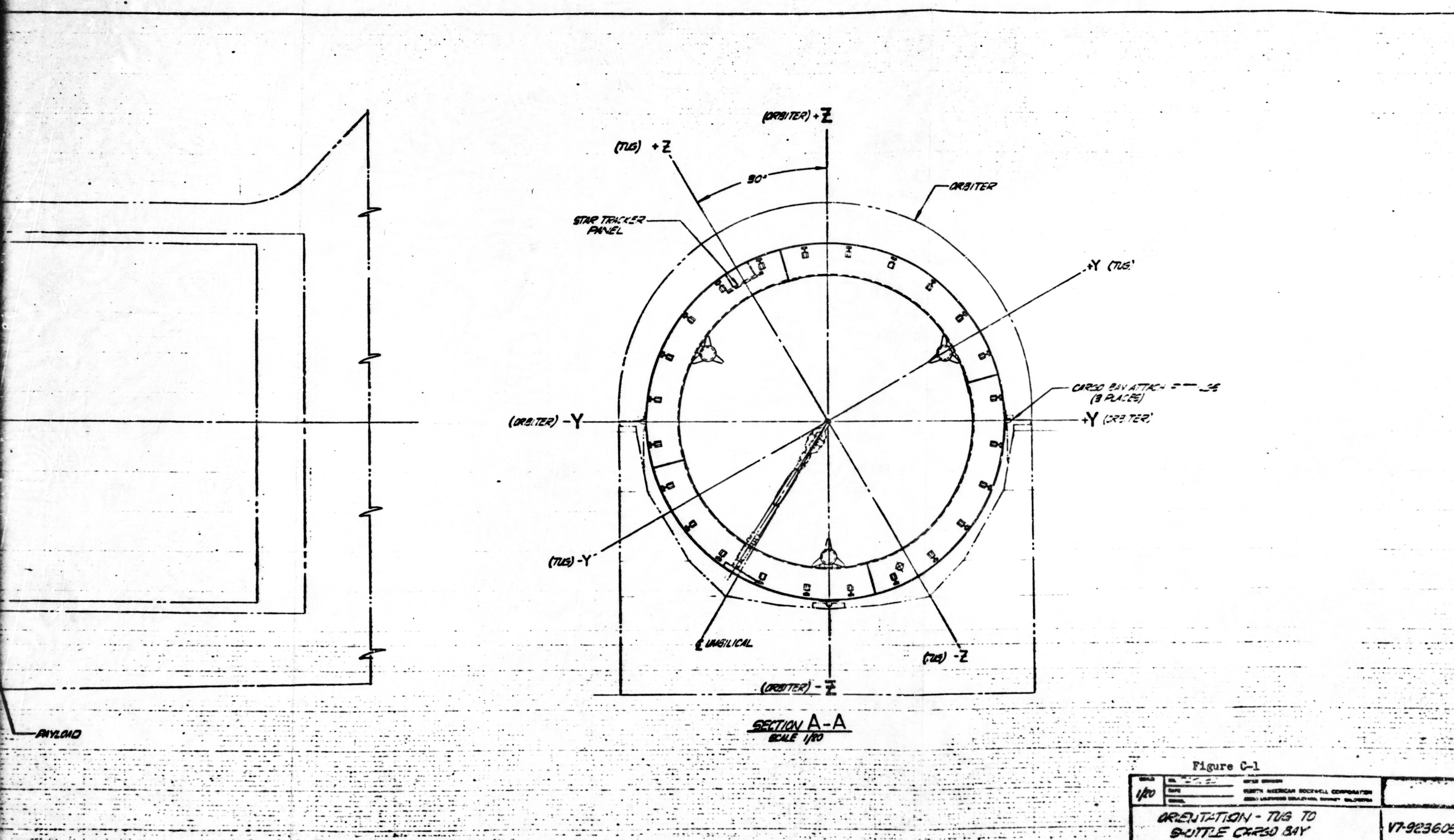

(ORBITER) +Z
(TUG) +Z
30°
ORBITER
STAR TRACKER PANEL
+Y (TUG)
CARGO BAY ATTACH
(3 PLACES)
(ORBITER) -Y
+Y (ORBITER)
(TUG) -Y
℄ UMBILICAL
(TUG) -Z
(ORBITER) -Z
PAYLOAD
SECTION A-A
SCALE 1/20
1/20
NORTH AMERICAN ROCKWELL CORPORATION
ORIENTATION - TUG TO SHUTTLE CARGO BAY
V7-923605

Figure C-1

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4.2.5 Factor of Safety

All primary and secondary structural components, where critical load conditions occur with the Space Tug attached, shall be designed with Safety factors of 1.4 for ultimate and 1.1 for yield. For structural components whose critical load condition occurs during Tug operations or other times where failure of the component has no adverse effect on the Shuttle, the safety factors of 1.25 times ultimate and 1.05 times yield shall be used. All structural components shall be designed for a 20 mission life.

4.3 Electrical Requirements

4.3.1 Connector Definition

The electrical interface between the Tug and the Shuttle shall be provided by means of electrical connectors. The functions of the electrical hardline connector shall be to provide power to the Tug during the mated ascent phase and for abort purposes. The connector also provides the link for updating the Tug navition equipment prior to Shuttle/Tug separation.

4.3.1.1 Connector Characteristics

The characteristics of each interface connector shall include the following:

a. Connector Type

b. Pin Assignments

c. Signal Function (Command/Measurement)

d. Signal Characteristics (Analog/Discrete/Voltage/Load/Waveforms)

4.3.2 Confidence Loop

A circulating confidence loop shall be provided through all connectors in series to provide an indication to the Shuttle or GSE that the connectors are properly mated.

4.4 Flight Criteria

4.4.1 Acceleration

Maximum axial acceleration of the mated Shuttle/Tug shall be as defined in Table C-2.

Table C-2. Tug Load Factors

Condition		X(g)	Y(g)	Z(g)
Launch		1.4 $\pm$1.6	$\pm$1.0	$\pm$1.0
High Q Booster Thrust High Q Booster Thrust		1.9 $\pm$0.3	$\pm$1.0	0.8 $\pm$0.2
End Boost (Booster Thrust)*	3	$\pm$0.3	$\pm$0.6	$\pm$0.6
End Burn (Orbiter Thrust)	3	$\pm$0.3	$\pm$0.5	$\pm$0.5
Orbiter Entry		-0.5	$\pm$1.0	-3.0 $\pm$1.0
Orbiter Flyback		-0.5	$\pm$1.0	$\pm$1.0 -2.5 $\pm$1.0
Landing		-1.3	$\pm$0.5	-2.7 $\pm$0.5

*Excludes booster-orbiter separation loads which are TBD

4.4.2 Tug Mission

All Tug missions shall be initiated from and terminated at the Shuttle in a 100 n.m. circular orbit inclined at 28.5 degrees.

4.5 Separation Criteria

Separation of the Tug from the Shuttle shall occur at the structural attachment interface upon command from the Shuttle. Prior to the separation command, the Tug will be provided a navigation update from the Shuttle.

4.5.1 Tug Deployment

Tug deployment shall occur after the Shuttle has been injected into the 100 n.m. circular parking orbit. The Tug/Shuttle docking mechanism and hard points for forward frame and possibly for Shuttle remote manipulators make up the Tug/Shuttle structural and deployment interfaces.

4.5.2 Design Criteria

4.5.2.1 Structural Components

All structural components of the separation subsystem shall be capable of withstanding loads imposed during ground operations and mated flight.

4.5.3 Engine Exhaust Plume Impingement

The design of the separation subsystem and its corresponding mode of deployment must consider the possibility of plume impingement on the Shuttle structure from the main engine burn of the Tug. The characteristics of the main engine plume are shown in Figure C-2.

4.5.4 Separation Clearance

The separation clearance requirements are as specified in Figures TBD and TBD.

4.5.5 Separation Sequence

Normal separation of the Tug from the Shuttle shall be accomplished in the timeline shown in Figures TBD* and TBD*. The deployment sequence of the Tug begins with the proper orientation of the Shuttle/Tug, followed with the opening of the cargo bay (the cargo bay may be opened any time after parking orbit insertion) and extension of the deployment mechanism. After the Tug is satisfactorily checked out, (including pre-start functions of the main propulsion system), it is uncoupled from the Shuttle cargo bay and a 300 meter separation between the Shuttle and Tug is maintained. Following the satisfactory check of the main propulsion subsystem, the Tug is readied for injection into the outbound transfer orbit.

4.6 Abort Requirements

The Shuttle computer shall be the command center for all abort decisions involving the Shuttle and the Tug. The command center shall monitor the Shuttle and Tug failures, generate automatic and manual abort signals, and control automatic abort initiation commands. Abort initiation shall be provided for the following conditions:

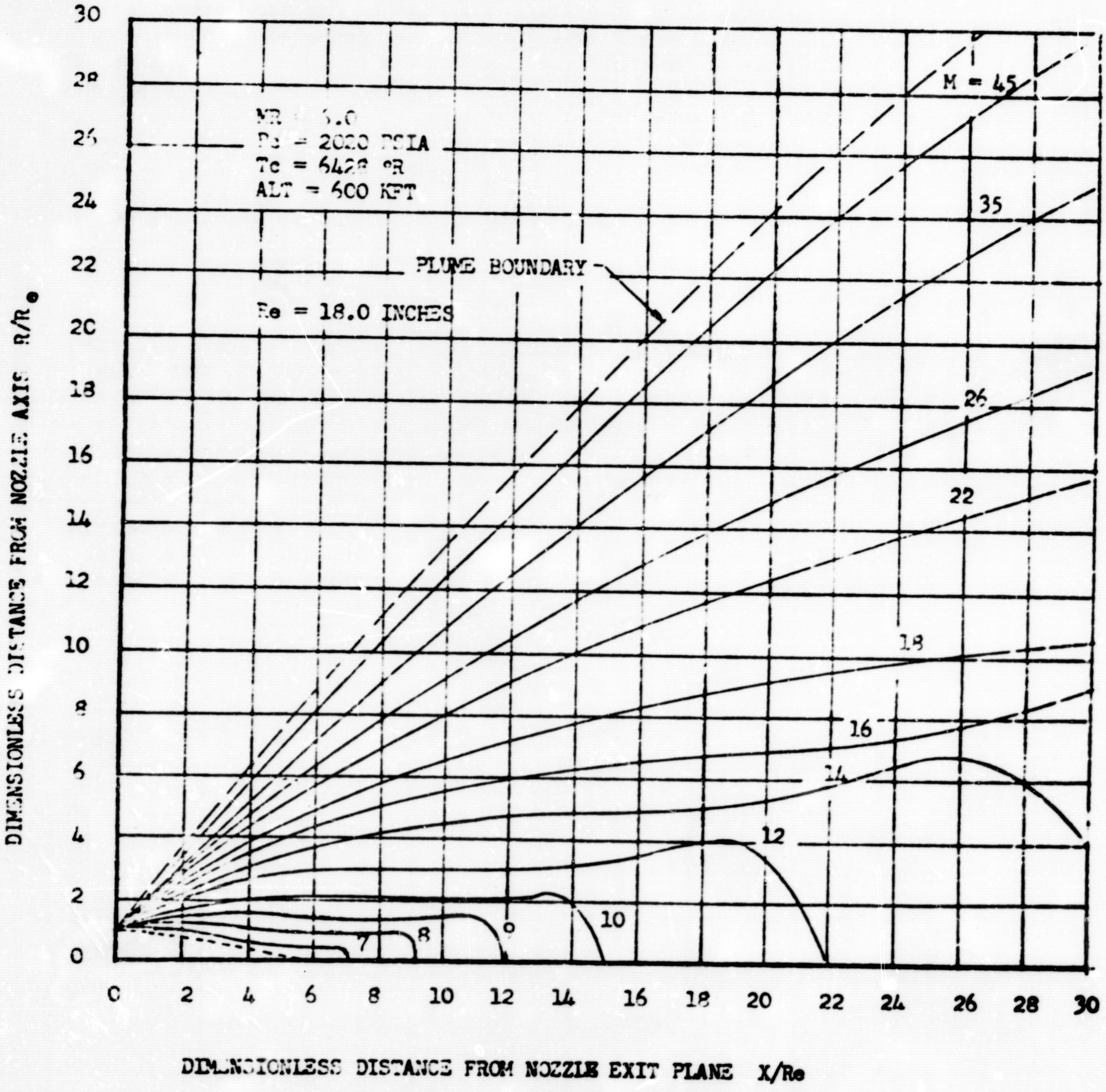
a. Manual abort (e.g., loss of critical subsystem to fail-safe level) if time permits; otherwise, the computer shall provide an automatic (back-up) abort.

b. Automatic abort for incapacitated crew.

*Similar to Figures 2.11-1, 2.11-3 & 2.11-4 of Volume III.

Figure C-2

SPACE SHUTTLE TUG
MAIN ENGINE EXHAUST PLUME ISOMACHS



4.6.1 Control

If mission readiness of the Tug to be injected into the transfer orbit is not satisfactorily reverified, the Tug with its payload must redock with the Shuttle and proceed with the abort procedure. Control of the maneuvers for retrieving the Tug by the Shuttle shall be provided from the Shuttle command center.

4.7 Environmental Criteria

4.7.1 Natural Environment

The Tug shall be designed to operate with the environmental limits as specified in NASA TMX53872, for the terrestrial environment and NASA TMX53957 for the space environment.

4.7.2 Induced Environment

The induced environments as defined in Tables C-2, C-3, C-4, and C-5 shall govern the design of the Tug.

Table C-3. Temperature Limits for the Internal Walls of the Cargo Bay

CONDITION	TEMPERATURES, °F MINIMUM	MAXIMUM
Pre-launch	-100	+120
Launch	-100	+200
On-orbit (door closed)	-100	+200
On-orbit (door open)	--	--
Entry and post landing	-100	+200

Table C-4. Orbiter Payload Compartment Internal Acoustic Design Criteria (Sound Pressure Level (db) Ref. 10-5 N/M^2)

1/3 OCTAVE CENTER BAND FREQ (HZ)	LIFT-OFF	BOUNDARY LAYER
5	124	124.5
6.3	127	125.0
8	128	126.0
10	129	126.5
12.5	131	127.0
16	132	128.0
20	134	128.5
25	135	129.0
31.5	137	130.0
40	138	130.5
50	139	131.0
63	140	132.0
80	141	132.5
100	143	133.0
125	144	134.0
160	145	134.5
200	145	135.5
250	145	136.0
315	144	136.5
400	143	137.0
500	142	137.5
630	141	138.0
800	140	138.5
1K	139	138.0
1.25K	138	137.0
1.6K	137	136.5
2K	135	135.5
2.5K	134	134.5
3.15K	133	134.0
4K	132	133.0
5K	131	132.0
6.3K	130	131.0
8K	129	130.0
10K	128	129.0
	OASPL155db	OASPL149db

Table C-5. Vibration Limits

Vibrations	
(1) Vehicle Dynamics	
Longitudinal axis (3-35 Hz at 3 Oct/Min)	TBD
Lateral axis (3-35 Hz at 3 Oct/Min)	TBD
(2) Liftoff Random Vibration	
20-2000 Hz	TBD
Time	TBD
(3) Boost Random Vibration	
20-2000 Hz	TBD
Time	TBD
(4) Shock Spectrum	
20-100 Hz	TBD

5.0 PHYSICAL INTERFACE REQUIREMENTS

The physical interfaces between the Shuttle and the Tug are shown in Interface Control Drawings* (similar to V7-941514).

*Figures 2.11-1, 2.11-3 & 2.11-4 of Volume III.

Space Division
North American Rockwell

SPACE TUG/MAIN ENGINE

INTERFACE CONTROL DOCUMENT

SPACE TUG/MAIN ENGINE

INTERFACE CONTROL DOCUMENT

Contents

1.0 SCOPE

This document specifies the interface requirements between the Space Tug and the main LOX/LH_2 engine. The requirements delineated herein include the following:

a. Propellant Feed System

b. Structural Load Limitation

c. Engine Control

d. Gimbal Mounts

e. Electrical Power

f. Gaseous Supply for Engine Purge

g. Pressure Measurements

2.0 APPLICABLE DOCUMENTS

The following documents, of the issue in effect at the time of the contract agreement, form a part of this document to the extent specified herein:

________________	Engines, rocket, liquid, propellant, general specifications for
MSFC-SPEC-356	Propellant, Liquid Hydrogen
MSFC-SPEC-399 Type II, Grade C	Propellant, Liquid Oxygen
MSFC-SPEC-364A	Helium
________________	Interface Control Drawing Tug/Engine
NASA TMX53872 1969 Revision	Terrential Environment Criteria Guidelines for use in Space Vehicle Development

3.0 ABBREVIATIONS AND ACRONYMS

LOX	Liquid Oxygen
LH_2	Liquid Hydrogen
TVC	Thrust Vector Control
TBD	To Be Determined
°F	Degree Farenheit

Fx	Force, x-axis
Fy	Force, y-axis
lbf	pounds force
lbm	pounds mass
Hz	Hertz
Mx	Moment about the x-axis
My	Moment about the y-axis
Mz	Moment about the z-axis
Oct	Octave
Min	Minute
X(g)	g-force in x-axis
y(g)	g-force in y-axis
z(g)	g-force in z-axis

4.0 FUNCTIONAL REQUIREMENTS

The functional requirements of the Space Tug/Main Engine interfaces shall be as specified in the following paragraphs.

4.1 Electrical Interface

4.1.1 Connector Definition

The electrical interface connectors necessary to provide the electrical power and commands to the engine from the Tug shall be as defined in Table C-6.

4.1.1.1 Connector Characteristics

The design and performance of each interface connector shall be characterized by the following parameters. Each connector shall be keyed to mate with only the proper half.

a. Connector Type

b. Pin Assignments

c. Signal Functions (commands/measurements)

d. Signal Characteristics (analog/discrete/voltage/waveform/load.

4.1.2 Power Requirements

The interface electrical power requirements shall be as defined in Table C-7. The values specified represent the maximum operating power requirements of the engine and the minimum output of the power supply.

Table C-6. Tug/Engine Electrical Connector Reference Designations

Engine Connector Number	Tug Connector Number	Function

Table C-7. Engine Electrical Requirements

Function	Power (Watts)			
	Peak	Continuous Operating	Shutdown Purge-5 Sec	Coast
Electronic Control (1)	200	200	200	10
Ignition (4)	360	-	-	-
Solenoid Valves (5)	940	125	100	-
Modulating Valves (4)	388	388	-	-
Gimbal Actuator				
Actuation	900	540	-	-
Heating	200	70	-	-

4.1.3 Shield Terminations

All shields shall terminate in the Tug.

4.1.4 Electrical Wiring

All electrical wiring shall be identified with wire number, system identification, wire size and potential.

4.1.5 Cryogenic Temperature Effect

Exposure of the electrical components to the cryogenic temperatures shall preclude the ingestion of moisture and salts at the ambient environment.

4.2 Environmental Criteria

4.2.1 Natural Environment

The Tug shall be designed to withstand the climatic and space environments as specified in NASA TMX53872, Terrential Environment (Climatic) Criteria Guidelines for use in Space Vehicle Development, 1969 Revision.

4.2.2 Induced Environment

The induced environments in which the Tug will be subjected to include external loads resulting from engine burns, thermal, acoustic, and vibrations. The loads imparted to the Tug are defined in Table C-8. The thermal environment of the Tug is defined in Table C-9. The acoustic environment is shown in Table C-10. The vibration limits are shown in Table C-11.

Table C-8. Tug Load Factors

CONDITION		X(g)	Y(g)	Z(g)
LAUNCH		1.4 ±1.6	±1.0	±1.0
HIGH Q BOOSTER THRUST		1.9	±1.0	0.8
HIGH Q BOOSTER THRUST		±0.3		±0.2
END BOOST (BOOSTER THRUST)*	3	±0.3	±0.6	±0.6
END BURN (ORBITER THRUST)	3	±0.3	±0.5	±0.5
ORBITER ENTRY		-0.5	±1.0	-3.0 ±1.0
ORBITER FLYBACK		-0.5	±1.0	+1.0 -2.5 ±1.0
LANDING		-1.3	±0.5	-2.7 ±0.5

*EXCLUDES BOOSTER-ORBITER SEPARATION LOADS WHICH ARE TBD

Table C-9. Temperature Limits for the Internal Walls of the Cargo Bay

CONDITION	TEMPERATURES, °F MINIMUM	MAXIMUM
Pre-launch	-100	+120
Launch	-100	+200
On-orbit (door closed)	-100	+200
On-orbit (door open)	--	--
Entry and post landing	-100	+200

Table C-10. Orbiter Payload Compartment Internal Acoustic Design Criteria (Sound Pressure Level (db) Ref. 10-5 N/M^2)

1/3 OCTAVE CENTER BAND FREQ. (HZ)	LIFT-OFF	BOUNDARY LAYER
5	124	124.5
6.3	127	125.0
8	128	126.0
10	129	126.5
12.5	131	127.0
16	132	128.0
20	134	128.5
25	135	129.0
31.5	137	130.0
40	138	130.5
50	139	131.0
63	140	132.0
80	141	132.5
100	143	133.0
125	144	134.0
160	145	134.5
200	145	135.5
250	145	136.0
315	144	136.5
400	143	137.0
500	142	137.5
630	141	138.0
800	140	138.5
1K	139	138.0
1.25K	138	137.0
1.6K	137	136.5
2K	135	135.5
2.5K	134	134.5
3.15K	133	134.0
4K	132	133.0
5K	131	132.0
6.3K	130	131.0
8K	129	130.0
10K	128	129.0
	OASPL155db	OASPL149db

Table C-11. Vibration Limits

Vibrations	
(1) Vehicle Dynamics	
Longitudinal axis (3-35 Hz at 3 Oct/Min)	TBD
Lateral axis (3-35 Hz at 3 Oct/Min)	TBD
(2) Liftoff Random Vibration	
20-2000 Hz	TBD
Time	
(3) Boost Random Vibration	
20-2000 Hz	TBD
Time	TBD
(4) Shock Spectrum	
20-100 Hz	TBD

4.3 Loads Requirements

4.3.1 Loads and Moments

The maximum allowable loads and moments for the specified interface areas are shown in Tables C-12 and C-13. The sign convention and coordinates shall be as specified in Figure C-3.

Table C-12. Allowable Structural Load Limits

	Axis	Gimbal Joint	Oxidizer Pump Inlet	Fuel Pump Inlet	Gimbal Actuator No. 1	Gimbal Actuator No. 2
Deflection (along axis shown)	X Y Z	TBD	TBD	TBD		
Rotation (about axis shown)	X Y Z	TBD	TBD			
Spring Rate (lb/in.)		TBD	TBD	TBD	TBD	TBD

When Integrating Nose to Tail

+ Force Produces + Shear
+Y Shear Produces + Bending Moment About Z
+Z Shear Produces - Bending Moment About Y

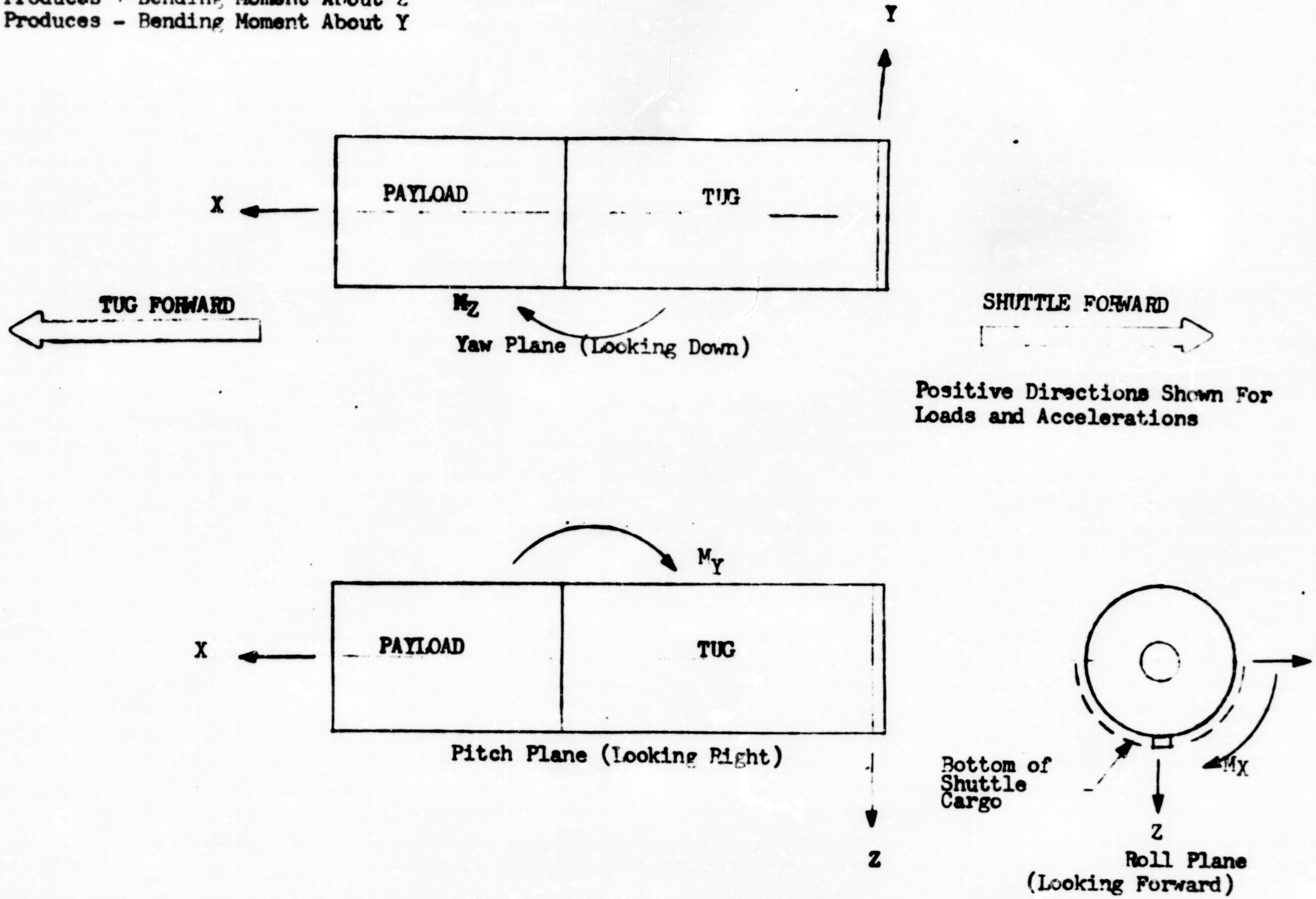


Figure C-3. LOADS SIGN CONVENTION

Table C-13. Engine Allowable Interface Loads

Maximum Allowable Loads at Gimbal Interface	Fx Fy Fz Mx My Mz	TBD TBD TBD TBD TBD TBD
Maximum Allowable Loads at Actuator #1 Vehicle Interface	Fx Fy Fz Mx My Mz	TBD TBD TBD TBD TBD TBD
Maximum Allowable Loads at Actuator #2 Vehicle Interface	Fx Fy Fz Mx My Mz	TBD TBD TBD TBD TBD TBD
Maximum Allowable Loads at Low Pressure Fuel Turbopump Inlet Interface	Fx Fy Fz Mx My Mz	TBD
Maximum Allowable Loads at Low Pressure Oxidizer Turbopump Inlet Interface	Fx Fy Fz Mx My Mz	TBD

NOTE: (1) Forces are in pounds.
(2) Moment are in inch-pounds.

4.4 Fluid Requirements

The inlet pressure range, flow rate and temperature limits at the interface between the Tug and the main engine are specified in Table C-14.

Table C-14. Fluid Characteristics

Description	Fluid Type	Pressure, PSIA			Nominal Flow Rate	Fluid Temperature Limits,	
		Min.	Nom.	Max.		Minimum	Maximum
Eng. Ox. Inlet	LOX	15	-	22.9	18.25 lb/sec	162.7°R	165.6°R
Eng. Fuel Inlet	LH_2	16	-	22.1	3.04 lb/sec	37.0°R	37.9°R
Engine Purge	Helium	750	0	-	-	350°R	
LH_2 Tank Press	GH_2	2000 - 3000			0.01 lb/sec	260°R	
LOX Tank Press	GO_2	2000 - 4000			0.05 lb/sec	600°R	

4.4.1 Media

The media that flow through the interface shall be as specified in Table C-15.

4.5 Propellant Feed Lines

4.5.1 Line Identification

Fluid lines shall be clearly marked to identify the fluid type, pressure range and direction of flow.

4.5.2 Line Routing

Fluid lines shall be routed in such a manner as to avoid sharp bends, protuberances and moving surfaces that could lead to leakage failure.

4.5.3 Safety Factors

The propellant feed lines (LOX and LH_2) shall be designed to adhere to the safety factors specified below:

Shuttle Operation
- Proof Pressure - 1.05
- Burst Pressure - 1.40

Tug Operation
- Proof Pressure - 1.05
- Burst Pressure - 1.25

4.6 Main Engine Requirements

4.6.1 Engine Characteristics

The performance and physical characteristics of the main engine are shown in Table C-16.

Table C-15. Engine Operating Fluids and Cleanliness Limits

Type	Maximum Particle Size, or Requirement [1]		Remarks
	Particle Size (x), Microns	Particles Allowable (No.)	
Helium, [2] MSFC-SPEC-364 or MIL-P-27407	x ≤30 30< x ≤100 x >100	No limit 25 0	
Liquid Oxygen, [3] MIL-P-25508	x ≤100 100< x ≤200 200< x ≤250 x >250	No limit 1000 500 0	Acetylene content shall be no larger than 1.55 ppm, soluble hydrocarbon shall not exceed 75 ppm, the purity not to be less than 99.2 percent, and the particulate content of the oxygen must not be limited by the total weight.
Liquid Hydrogen, [3] MIL-P-27201	x ≤100 100< x ≤200 200< x ≤250 x >250	No limit 1000 500 0	

NOTES:

[1] Cleanliness limits specified are the maximum allowable at the engine-to-vehicle interface.

[2] Maximum number of particles based on a 30 standard cubic foot sample.

[3] Maximum number of particles based on a 100 ml sample.

Table C-16. Main Engine Characteristics

Number of Engines	One (1)
Propellants	Liquid Oxygen/Liquid Hydrogen
Minimum Vacuum Thrust, lbs.	10,000
Engine Length/Diameter, Inches (Max)	72/36.5
Vacuum Thrust Throttling Capability	5.0 to 1
Engine Weight, Lbs.	213
Number of Vacuum Starts	160
Service Life Between Overhauls (reuseable mode), thermocycles	300
Service Life Between Overhauls (reuseable mode), hours	20
Gimbal Angle (square pattern), degrees	± 7
Gimbal Acceleration, radials/second^2	20
Minimum natural Frequency of Gimbal System, Hertz	10
Fuel Pump NPSH, Feet of Hydrogen	15
Oxidizer Pump NPSH, feet of oxygen	2
Maximum Single run Duration, Seconds	1400
Maximum Time Between Firings (coast), days	3
Minimum Time Between Firings (coast), minutes	10
Service-free Engine Run Time, hours	1
Service-free Engine Firing Cycles	10
Max. Storage Time in Orbit (Dry), Weeks	1

4.6.2 Engine Purge

The engine compartment shall be capable of being purged by conditioned gas while in the shuttle cargo bay on the launch pad.

4.6.2.1 Purge Gas

Helium gas or MSFC-SPEC-364 or MIL-P-27407 shall be used for the engine purge.

4.6.3 Thrust Vector Control

Thrust vector control in pitch and yaw shall be provided by the use of two (2) gimbal actuators.

4.6.3.1 Gimbal Capability

The engine gimballing capability shall provide a thrust vector making an angle, in a square pattern, of ± 7.0 degrees from the

engine geometrical centerline (Figure C-4). The nominal gimbal angular acceleration shall be 20 radials per second squared. Other gimbal actuator data are shown in Table C-17.

Table C-17. Tug Thrust Vector Control

Gimbal Angle -	±7 Deg. Sq. Pattern
Gimbal Velocity at Design Load - - - - - - - - - - - - -	8 Deg./Sec.
Actuator-to-Engine Moment Arm (at Actuator Null) - -	12 In.
Actuator Stroke -	1 ± 1.5 In.
Actuator Stroke -	20 In.
Actuator Stall Force - - - - - - - - - - - - - - - - - - -	1800 Lb.
Gimbal Acceleration -	12 Deg./Sec^2
Total Wet (Including Motor but not Cabling & Inverter) - - - - - - - - - - - - -	40 Lb.

4.6.3.2 Gimbal Mounts

Mounting of the gimbal actuators shall be as shown in Interface Control Drawing*(Similar to V7-941512).

5.0 PHYSICAL INTERFACE REQUIREMENTS

The physical interface requirements between the Tug and the main engine proper are shown in Interface Control Drawing*_____. The Tug main engine schematic is shown in Figure C-5.

*Figure 2.10-2 of Volume III

*Similar to Figures 2.8-3 of Volume III

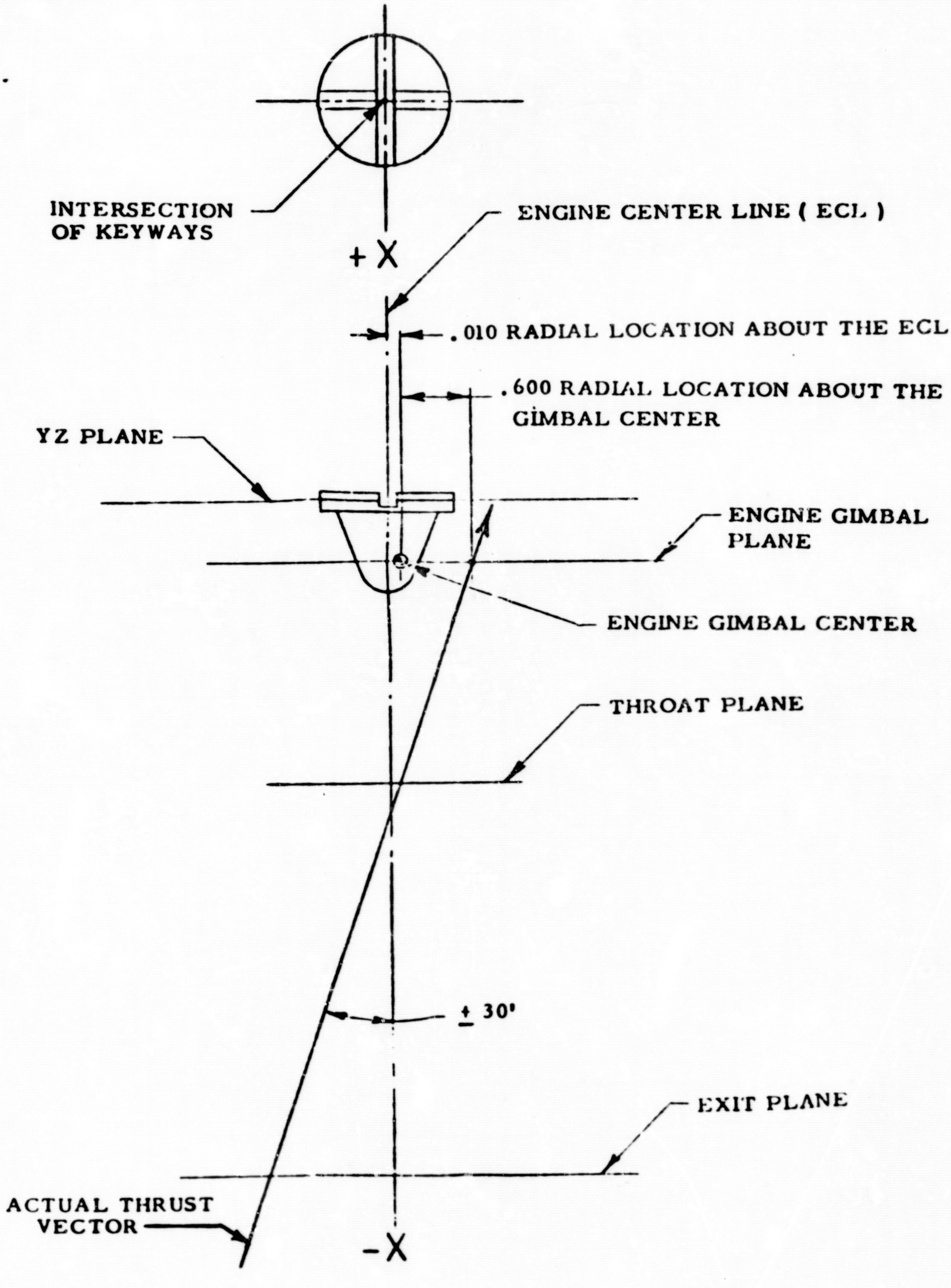


Figure C-4. Thrust Vector Definition

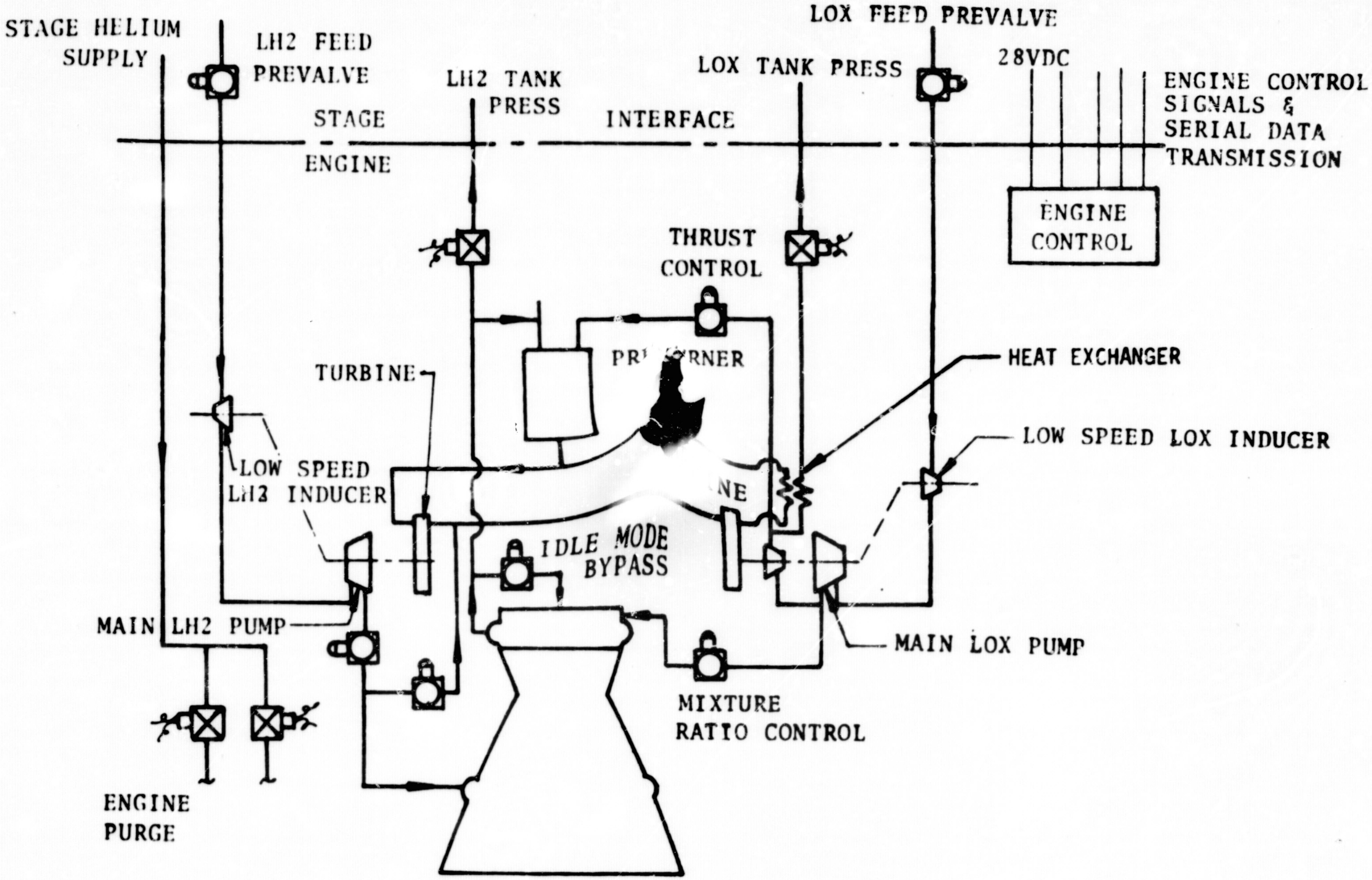


Figure C-5. Tug Main Engine Schematic

SPACE TUG/PAYLOAD
INTERFACE REQUIREMENTS

Space Division
North American Rockwell

INTERFACE CONTROL DOCUMENT

Table of Contents

1.0 SCOPE

This document specifies the functional, physical and procedural interfaces of the Space Tug to its payload. It defines the requirements and criteria to be observed in the design of interfacing equipment.

2.0 APPLICABLE DOCUMENTS

2.1 Specifications

TBD

2.2 Interface Control Documents

TBD

2.3 Drawings

TBD

2.4 Manuals and Handbooks

TBD

3.0 ABBREVIATIONS AND SYMBOLS

db	decibel
Deg.	Degrees
ft	feet/foot
freq	frequency
Hz	Hertz
ICD	Interface Control Document/Drawing
Max	Maximum
Min	Minimum
Oct	Octave
sec	second
TBD	To Be Determined
TV	Television

4.0 FUNCTIONAL REQUIREMENTS

The Space Tug/Payload functional interfaces define the structural limits, the environmental, separation and docking criteria as indicated by the following paragraphs.

4.1 Electrical Requirements

None

4.2 Structural Requirements

4.2.1 Structure

The structure of the Tug/Payload interface shall be capable of transferring the structural loads through the interface and shall be designed to meet or exceed the limit loads given in the following paragraphs using a minimum allowable safety factors of 1.4 ultimate and 1.1 yield where critical load conditions occur while the Space Tug/Payload is attached to the Space Shuttle. When failure of a structural component has no effect on the Space Shuttle System, the component may be designed with a minimum allowable safety factor of 1.25 ultimate and 1.05 yield during Tug operation. All structural components shall be designed for positive margins of safety for 20 mission cycles.

4.2.2 Space Tug/Payload Interface Load Limits

Condition	Shear (Lb)	Bending Moment (In. Lb X 10^{-6})	Longitudinal Compression (Lb)	Longitudinal Tension (Lb)
Launch	∓8,060	±1.22	1,600	24,200
High Q Booster Thrust		Non Critical		
End Boost (Booster Thrust)*	∓4,800	±0.73	--	26,600
End Burn (Orbiter Thrust)		Non Critical		
Orbiter Entry	-16,640	+2.53	2,080	--
Orbiter Flyback		Non Critical		
Landing	-13,310	+2.02	5,400	--

*Excludes Booster-Orbiter Separation Loads Which are TBD.
See Figure C-6 for loads sign convention.

4.2.3 Venting, Structural Leakage, Drainage

4.2.3.1 Venting

Venting of the Space Tug/Payload interstage area shall be accomplished by TBD square inches maximum opening(s) on the payload aft interstage.

When Integrating Nose to Tail

+ Force Produces + Shear
+Y Shear Produces + Bending Moment About Z
+Z Shear Produces - Bending Moment About Y

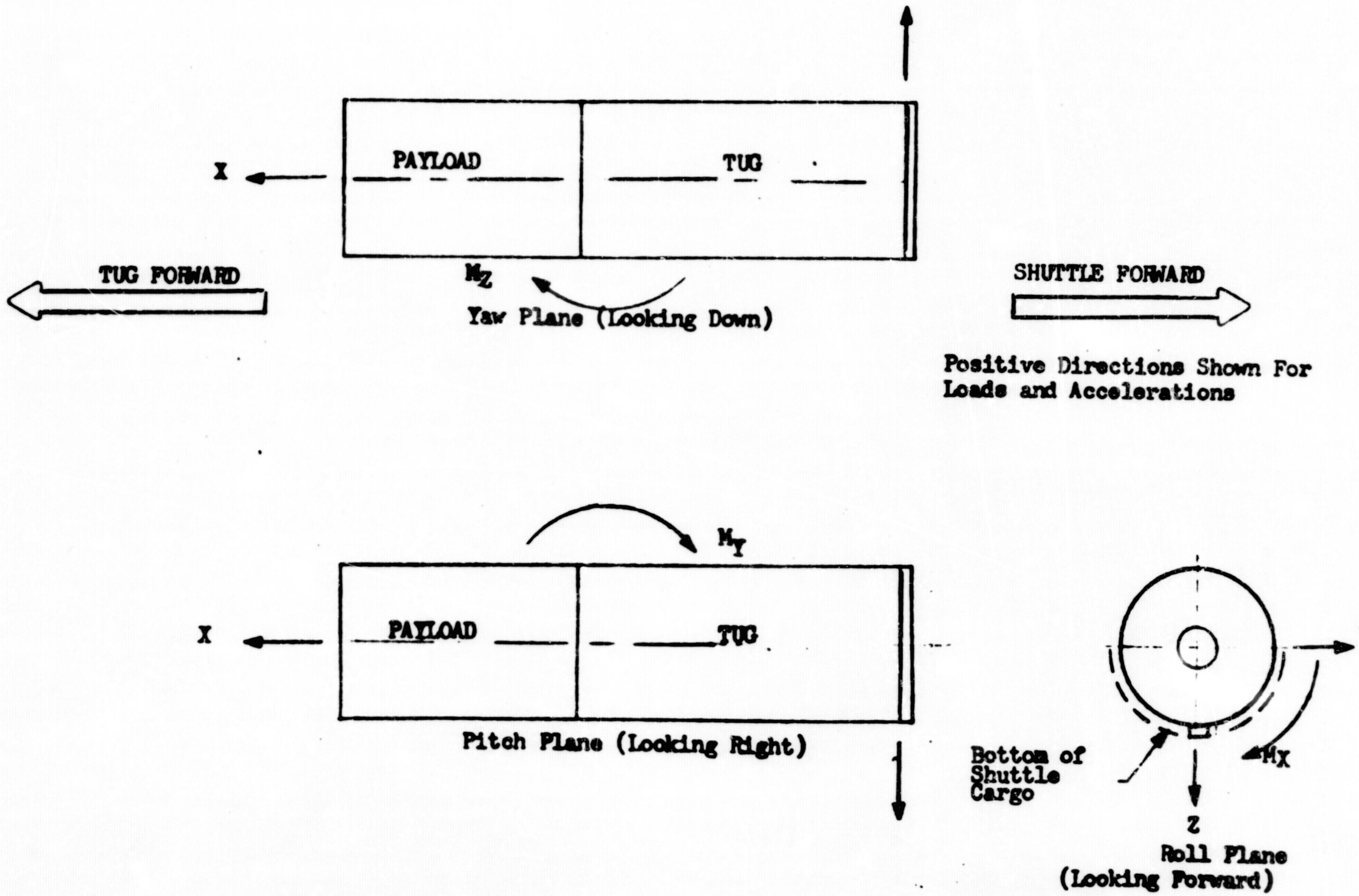


Figure C-6. LOADS SIGN CONVENTION

4.2.3.2 Structural Leakage

Structural leakage in the Space Tug/Payload interface area, in excess of that listed in Paragraph 4.2.3.1, shall be designed not to exceed the following:

a. In the Space Tug forward skirt area TBD square inches (equivalent).

b. In the Payload aft skirt area TBD square inches (equivalent).

4.2.3.3 Drainage

Drain holes in the Space Tug forward skirt area shall be plugged prior to launch.

4.3 Critical Environmental Parameters

4.3.1 Temperature

The temperature environment of the Shuttle cargo bay for periods shown are as follows:

Prelaunch	-100 to 120°F
Launch	-100 to 200°F
On-Orbit (door closed)	-100 to 200°F
On-Orbit (door open)	TBD
Entry and post landing	-100 to 200°F

Heating effects from protuberances as defined in ICD TBD shall be considered for Space Tug design.

4.3.2 Acoustic

See Table C-18.

4.3.3 Vibrations

a. Vehicle Dynamics

Longitudinal Axis (3-35 Hz at 3 Oct/Min)	TBD
Lateral Axis (3-35 Hz at 3 Oct/Min)	TBD

b. Liftoff Random Vibration

200 - 2000 Hz	TBD
Time	TBD

c. Boost Random Vibration

20 - 2000 Hz	TBD
Time	TBD

d. Shock Spectrum

20 - 100 Hz	TBD

TABLE C-18
ORBITER PAYLOAD COMPARTMENT
INTERNAL ACOUSTIC DESIGN CRITERIA
(SOUND PRESSURE LEVEL (db) REF. 10^{-5} N/M^2)

1/3 Octave Center Band Freq. (Hz)	Lift-Off	Boundary Layer
5	124	124.5
6.3	127	125.0
8	128	126.0
10	129	126.5
12.5	131	127.0
16	132	128.0
20	134	128.5
25	135	129.0
31.5	137	130.0
40	138	130.5
50	139	131.0
63	140	132.0
80	141	132.5
100	143	133.0
125	144	134.0
160	145	134.5
200	145	135.5
250	145	136.0
315	144	136.5
400	143	137.0
500	142	137.5
630	141	138.0
800	140	138.5
1K	139	138.0
1.25K	138	137.0
1.6K	137	136.5
2K	135	135.5
2.5K	134	134.5
3.15K	133	134.0
4K	132	133.0
5K	131	132.0
6.3K	130	131.0
8K	129	130.0
10K	128	129.0
	OASPL155db	OASPL149db

4.3.4 Pressure

(Pressurization requirements for Space Tug/Payload interstage area during Space Tug boost - if required).

TBD

4.4 Payload Deployment/Separation Parameters

The mode of separation used for the Space Tug/Payload shall be by means of mechanical screw jack actuated mechanisms. The Space Tug/Payload separation shall occur at Station 454.5 upon attainment of geosynchronous orbit.

4.4.1 Dynamic Separation Criteria

Initial conditions at physical separation:

1. Angle of Attack = TBD degrees (max) (If Applicable)

2. Attitude Error = TBD degrees (deviation from nominal vehicle attitude)

3. Attitude Rate = TBD degrees/sec. (max)

4.4.2 Separation Altitude

Separation altitude shall be as defined in documents TBD.

4.4.3 Space Tug Venting During Separation

Disturbance caused by venting of the Space Tug propellant tanks during Space Tug/Payload separation sequence shall be a maximum of TBD pounds.

4.4.4 Deployment/Separation Sequence

TBD

4.5 Rendezvous/Docking Parameters

The Space Tug guidance shall be capable of rendezvousing with the Payload to be retrieved. The Space Tug shall perform all translational and attitude maneuvers required to accomplish rendezvous to within 300 meters of Payload. The Space Tug to Payload docking accuracy requirements shall be as follows:

PARAMETERS	STRUCTURAL	G. AND C.
Centering Miss Distance	0 to 1.0 Foot	0 to 0.75 Foot
Miss Angle	0 to 5.0 Degrees	0 to 1.0 Degree
Longitudinal Velocity	0.1 to 1.0 Ft/Sec.	0 to 1.0 Ft/Sec.
Lateral Velocity	0 to 0.30 Ft/Sec.	0 to 0.3 Ft/Sec.
Angular Velocity (combined maximum of pitch, yaw and roll motion)	0 to 0.50 Deg./Sec.	0 to 0.50 Deg/Sec.

- Final docking shall be accomplished from ground control utilizing Space Tug onboard TV camera system.

4.6 Fluid Requirements (None)

5.0 PHYSICAL REQUIREMENTS

The Space Tug/Payload physical interface shall be in accordance with Figure C-7.

6.0 PROCEDURAL REQUIREMENTS

The following paragraphs specify the procedural requirements for Space Tug/Payload mating, de-mating, interface alignment and other services performed involving procedural interface requirement.

6.1 Accessibility

Access equipment provided for the Space Tug/Payload interface area shall be designed to support the required operations in the interface area. The access provisions shall be compatible with the following interface operations:

6.2 Space Tug/Payload Mating

(Procedures describing the mating of the Space Tug and Payload.)

TBD

6.3 Space Tug/Payload De-Mating

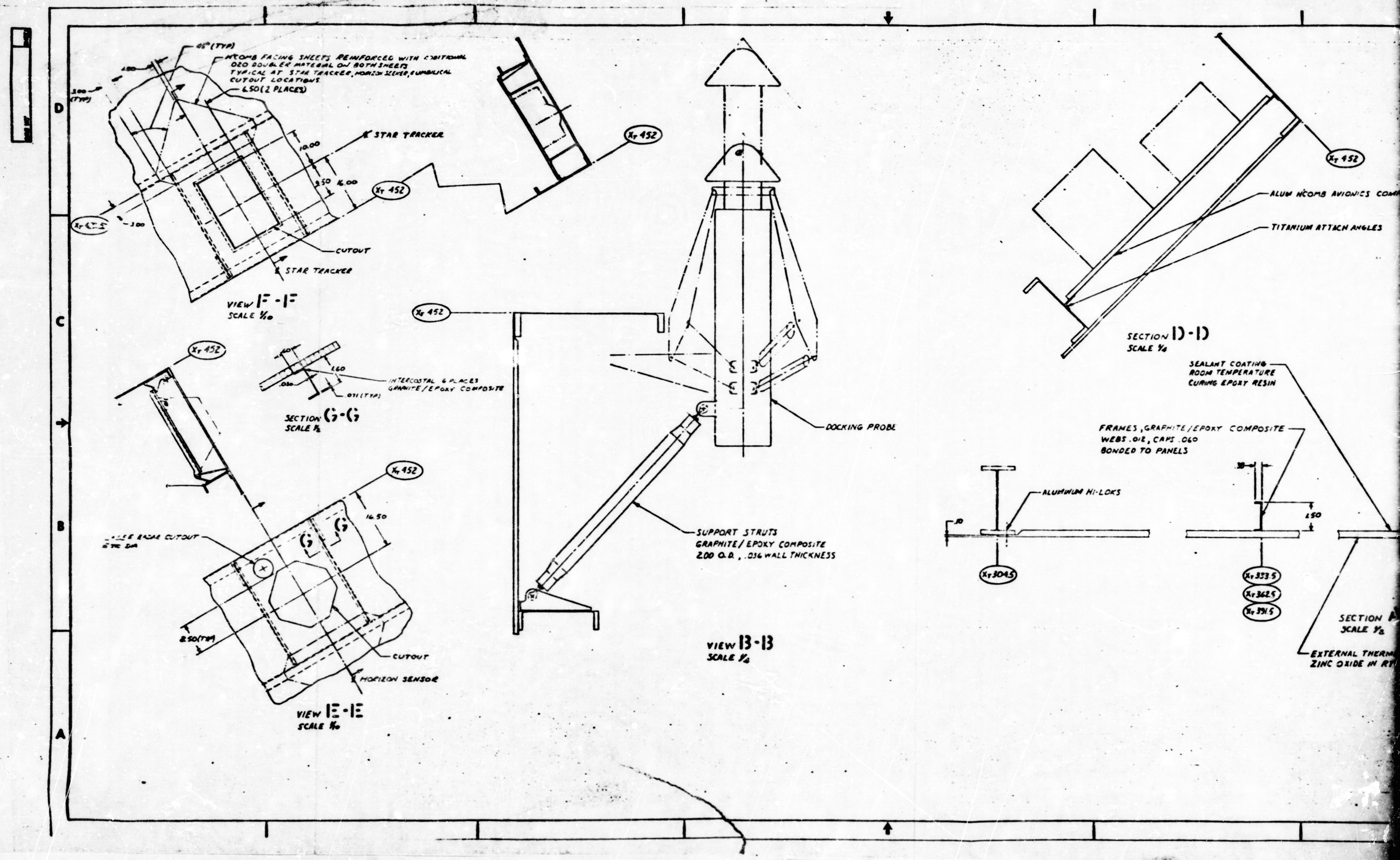
(Operations performed on the Space Tug or Payload that would require de-mating of the Payload from the Space Tug.)

TBD

6.4 Interface Alignment

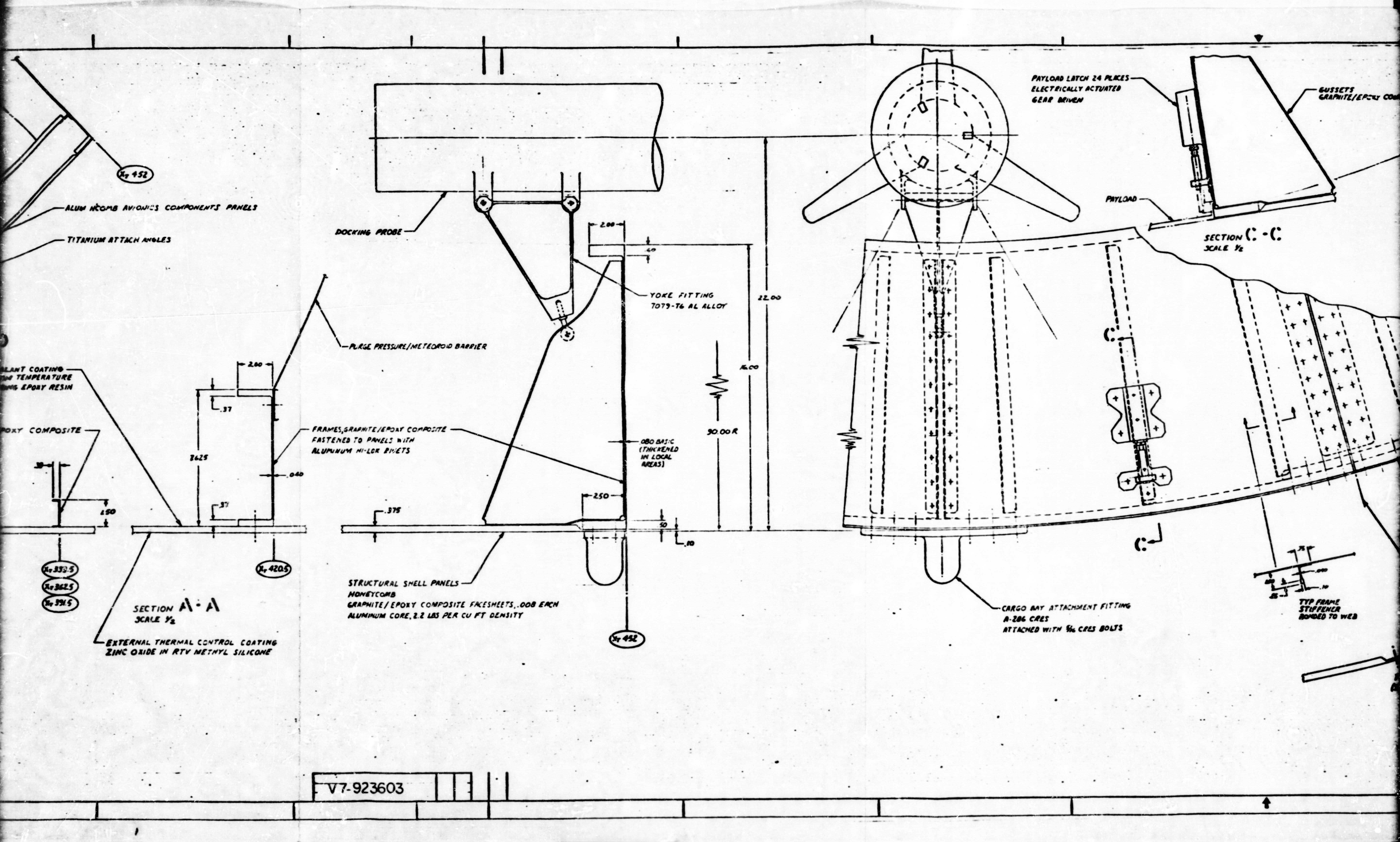
Alignment Procedures.

TBD

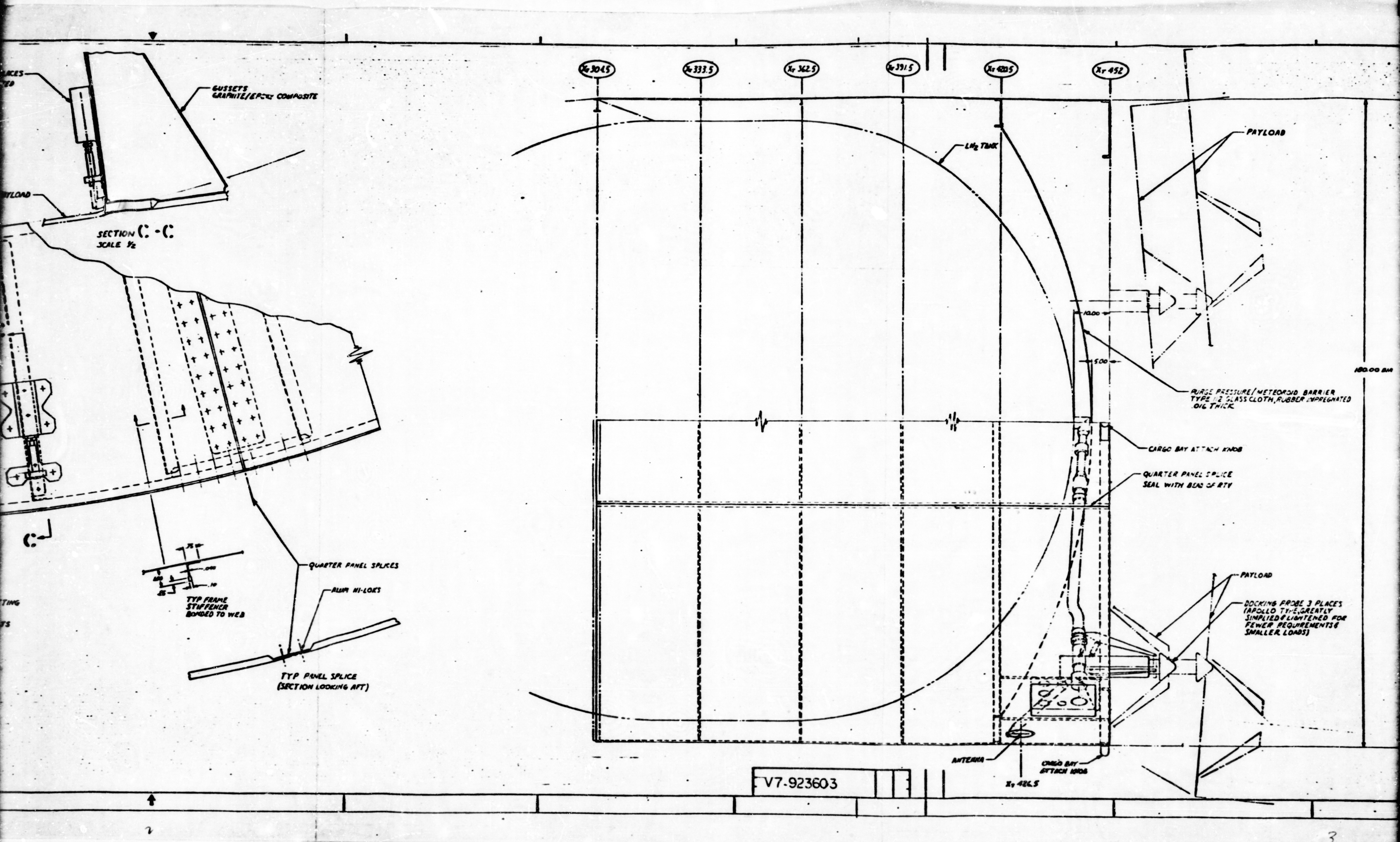


F-1

POOR FILMING QUALITY



F-2 POOR FILMING QUALITY


GUSSETS GRAPHITE/EPOXY COMPOSITE
SECTION C-C
SCALE 1/2
QUARTER PANEL SPLICES
TYP FRAME STIFFENER BONDED TO WEB
ALUM HI-LOKS
TYP PANEL SPLICE (SECTION LOOKING AFT)
X_T 304.5
X_T 333.5
X_T 362.5
X_T 391.5
X_T 420.5
X_T 452
LH_2 TANK
PAYLOAD
PURGE PRESSURE/METEOROID BARRIER TYPE 12 GLASS CLOTH, RUBBER IMPREGNATED .016 THICK
CARGO BAY ATTACH KNOB
QUARTER PANEL SPLICE SEAL WITH BEAD OF RTV
DOCKING PROBE 3 PLACES (APOLLO TYPE, GREATLY SIMPLIFIED & LIGHTENED FOR FEWER REQUIREMENTS & SMALLER LOADS)
ANTENNA
CARGO BAY ATTACH KNOB
X_T 426.5
180.00 DIA
V7-923603

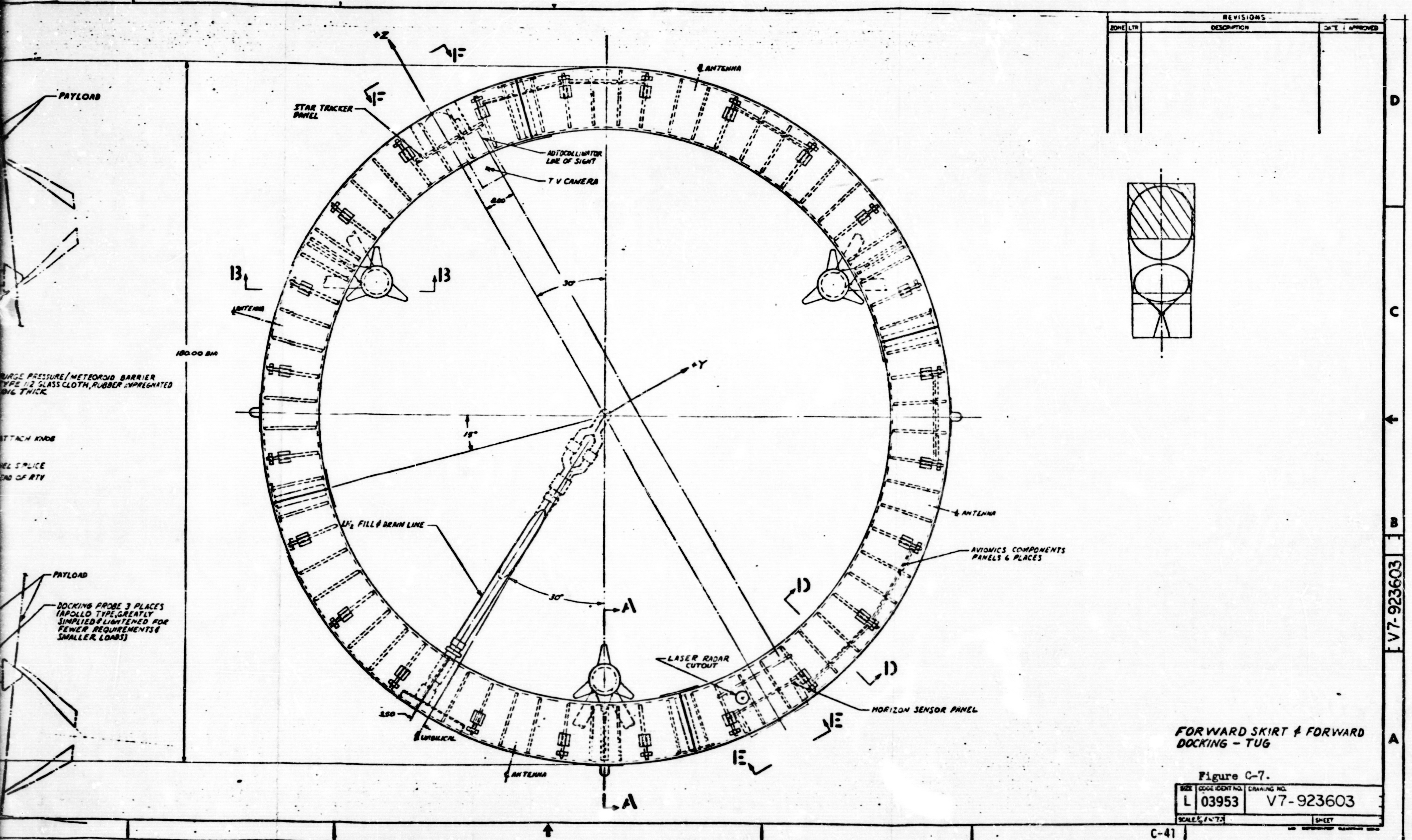


Figure C-7.

C-41

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TUG/GSE INTERFACE REQUIREMENTS

VIA SHUTTLE/GSE INTERFACE

TABLE OF CONTENTS

1.0 SCOPE

This document specifies the Physical and Functional requirements necessary at the Shuttle umbilical interfaces to support the services and maintenance of the Tug after Tug/Shuttle assembly and prior to launch.

2.0 APPLICABLE DOCUMENTS

2.1 Specifications

TBD

2.2 Drawings

TBD

2.3 Interface Control Documents

TBD

3.0 ABBREVIATIONS AND SYMBOLS

TBD

4.0 REQUIREMENTS

4.1 Media Specifications - The notes listed below give standard procurement specifications and media requirements at the interface.

4.1.1 Helium (He)

MSFC-SPEC-364

4.1.2 Nitrogen (N_2)

MSFC-SPEC-234

4.1.3 Hydrogen, Liquid (LH_2)

MSFC-SPEC-356

Propellant Hydrogen shall meet the following requirements at the interface when tested in accordance with the methods outlined in ______________________________ .

Purity and Impurity requirements - TBD

4.1.4 Oxygen Liquid (LOX)

MSFC-SPEC-399, Type II, Grade C

Propellant oxygen shall meet the following requirements at the interface when tested in accordance with the methods outlined in ____________________________.

Purity and Impurity requirements are as follows:

TBD

4.1.5 Hydraulic Fluid

MIL-H-5606

Maximum contamination level is as follows:

TBD

4.2 Fluids - Fluid Requirements are tabulated in Table C-19

4.2.1 Definitions of the Table C-19 headings are as follows:

Line Run Designation Number - An alpha-numerical sequence used to identify a function.

Connector Number - Reference designation of the equipment fitting that interfaces at the Shuttle and/or Tug interface.

Configuration Sheet and Zone also defined.

Functional Operation - A description of the purposes for the requirements.

Media - Identification of the fluid being transferred.

Pressure - The nominal fluid pressure and its tolerance or pressure range, expressed in pounds per square inch gage, required at the Shuttle interface.

Temperature - The fluid temperature and its tolerance expressed in degrees Rankine, at the Shuttle interface. The temperature stated is either that which is required or the existing ambient temperature, as applicable.

Quantity - The total weight of the fluid, in pounds, required to perform the function or the total weight required at liftoff.

TABLE C-19.

FLUID REQUIREMENTS

Line Run Desig No.	CONNECTOR & NO.				FUNCTIONAL OPERATIONS	REQUIREMENTS										Remarks
	SHUTTLE		TUG			Line Size	Media	Pressure (PSIG)		Temperature		Flowrate Lb/Min		Quantity (Lb)	Time (Min)	
	No.	Loc	No.	Loc				Min.	Max	Min.	Max.	Min.	Max.			
A.0	TBD	TBD			Fuel Transfer	3"	LH_2									
.1					Precool											
.2					Load			10					270	8100	30	
.3					Replenish											
.4					Drain											
.5					Preconditioning											
B.0					Oxidizer Transfer	3"	LOX									
.1					Precool											
.2					Load			15					1620	48,600	30	
.3					Replenish											
.4					Drain											
.5					Preconditioning											
C.0					GOX Accum. Tank Fill	½"	GOX		1250	400	±25		30	15	0.5	
					GOX Accum. Tank PUrge	½"	He		575	AMB	AMB					
D.0					GH_2 Accum. Tank Fill	½"	GH_2		1250	200	±25		6	3	0.5	
					" " " Purge	½"	He		575	AMB	AMB					
E.0					He Tank Fill	½"	He		3500	AMB	AMB	8	10	92	15	

SHEET NOTES

TABLE C-19 (continued)

FLUID REQUIREMENTS

Line Run Desig No.	CONNECTOR & NO. SHUTTLE No.	SHUTTLE Loc	TUG No.	TUG Loc	FUNCTIONAL OPERATIONS	REQUIREMENTS Line Size	Media	Pressure (PSIG) Min.	Pressure (PSIG) Max	Temperature Min.	Temperature Max	Flowrate Lb/Min Min.	Flowrate Lb/Min Max	Quantity (Lb)	Time (Min)	Remarks
H.0					GH_2 Vent	2"	GH_2	16 23	17 24		660 37		11.1	NA		
J.0					T/D Vent - Vac.	TBD	GH_2									
K.0					GOX Vent	3"	GOX	15 23	16 24		660		8.1	NA		
L.0					T/D Vent - Vac.	TBD	GOX									
M.0					LH_2 Tank Purge	TBD	H_e		3000	AMB	AMB	TBD	TBD	TBD	TBD	
N.0					LOX Tank Purge	TBD	H_e		3000	AMB	AMB	TBD	TBD	TBD	TBD	

SHEET NOTES

Time - The total time, in numbers, required to perform the normal functional operation.

Flowrate - The flowrate of the fluid, expressed in pounds per minute, at the Shuttle interface. All values for flow rates are expressed as nominal values except for pressurization functions which are expressed as maximum.

Remarks - Statements applicable to criteria that limit, clarify or identify interface parameters.

Line Size - Inside diameter of line at the Shuttle interface.

4.3 Electrical Requirements

TBD

4.4 Umbilical Requirements

4.4.1 Tug/Forward Umbilical Plate

Physical Configuration - See Table C-20.

Space Envelope
Plate Assembly
Attach Points to Tug
Connector Configuration

Functional Requirements - TBD

Actuation of Umbilical Plate
Restrictions

Environmental - TBD

Temperature
Humidity

4.4.2 Tug/Aft Umbilical Plate

Physical Configuration - See Table C-21.

Space Envelope
Plate Assembly
Attach Points to Tug
Connector Configuration

Functional Requirements - TBD

Actuation of Umbilical Plates
Restrictions

TABLE C-20.

TUG FORWARD UMBILICAL SERVICE CONNECTIONS

Connector Number	Dimensions			Connection Configuration	Remarks
	A	B	C		

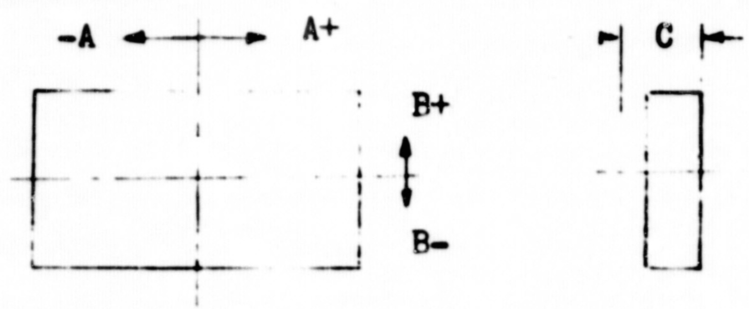
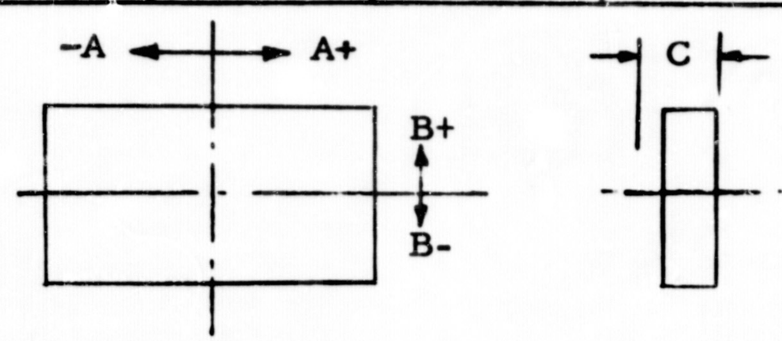


TABLE C-21.

TUG AFT UMBILICAL SERVICE CONNECTIONS

Connector Number	Dimensions			Connection Configuration	Remarks
	A	B	C		



Environmental - TBD

Temperature
Humidity

4.4.3 Shuttle/Umbilical Plate (Tug Fwd. and Aft)

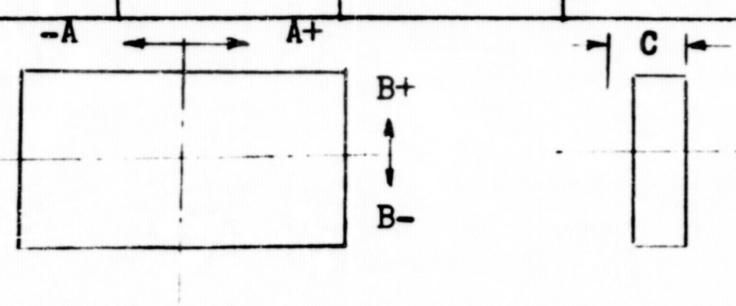
Physical - See Table C-22.

Connector Configuration
Plate Assembly

TABLE C-22.

SHUTTLE UMBILICAL SERVICE CONNECTIONS

Connector Number	Dimensions			Connector Configuration	Remarks
	A	B	C		



SPACE TUG/GROUND SUPPORT EQUIPMENT INTERFACE REQUIREMENTS

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

INTERFACE CONTROL DOCUMENT

TABLE OF CONTENTS

1.0 SCOPE

This document specifies the functional and physical interfaces of the Space Tug/Ground Support Equipment (GSE) during Space Tug checkout and service after completion of manufacture and during maintenance/refurbishment cycle prior to Space Tug/Payload combination stowage in the Shuttle Orbiter cargo bay. It defines the requirements and criteria to be observed in the design of interfacing equipment.

2.0 APPLICABLE DOCUMENTS

2.1 Specifications

MSFC-SPEC-364	Helium
MIL-P-27407	Propellant Pressurizing Agent, Helium
MIL-P-27201	Propellant, Hydrogen
MIL-P-27401	Propellant Pressurizing Agent, Nitrogen
MIL-P-25508	Propellant, Oxygen, Type II
MIL-H-5606	Hydraulic Fluid, Petroleum Base, Aircraft and Ordnance

2.2 Interface Control Documents

TBD

2.3 Drawings

TBD

2.4 Manuals and Handbooks

TBD

3.0 ABBREVIATIONS AND SYMBOLS

AWG	American Wire Gauge
GSE	Ground Support Equipment
He	Helium
ICD	Interface Control Document/Drawing
LH_2	Liquid Hydrogen
LOX	Liquid Oxygen
P.U.	Propellant Utilization
TBD	To Be Determined
W	Watts

4.0 FUNCTIONAL REQUIREMENTS

The functional requirements of the Space Tug/GSE interfaces shall be as defined in the following paragraphs.

4.1 Electrical Requirements

4.1.1 Connector Definition

The electrical interface of the Space Tug/GSE consist of the following functions:

A. Ground Electrical Power

B. Data Management

C. Critical Hardware Functions

D. P.U. Loading Computer

These electrical interfaces for the above functions shall consist of TBD pin connectors. The detailed characteristics of each connector shall be defined in subsequent phases of the Space Tug study.

4.1.2 Wire Size

The minimum wire size to be used shall be AWB No. TBD. Connector pins may be bussed together when current exceeds the capacity of a single pin is required.

4.1.3 Voltage Levels

Unless otherwise noted, the voltage level used in this document shall be 28 volts D.C.

4.1.4 Shield Termination

Unless otherwise noted, all shields shall terminate in the GSE. Shields shall be isolated from other shields by systems and functions within systems.

4.1.5 Power Requirements

The interface electrical power requirements shall be as defined in the table below. The values given are the maximum operating power requirements of the Space Tug and the minimum power requirements to the supplying GSE.

FUNCTION	REQUIREMENTS	REMARKS
CONTROL INSTRUMENTATION/COMMUNICATION	560W 480W	

4.2 Fluid Requirement

This section establishes the fluid requirements of the Space Tug at the interfaces of the umbilical disconnects. These requirements include all functions requiring transfer of fluids between the Space Tug and GSE during vehicle checkout and servicing.

4.2.1 Detailed Fluid Requirements

4.2.1.1 Propellants and Gases

Table C-23 defines the requirements for propellants and gases and the definitions of the headings are as follows:

- Item

  A number assigned to each functional operation for reference.

- Functional Operation

  The task being performed across the interface between the Space Tug and GSE.

- Media

  The fluid being transferred during functional operation.

- Pressure

  The nominal pressure and its tolerance or pressure range required at the interface.

- Temperature

  The temperature and its tolerance at the interface.

- Flowrate

  The flowrate of fluid at the interface.

TABLE C-23. PROPELLANTS AND GSE

ITEM	FUNCTIONAL OPERATION	MEDIA	FUNCTIONAL REQUIREMENTS					DISCONNECT REQUIREMENTS	
			PRESS (PSIG)	TEMP (°R)	FLOW (LB/MIN)	TIME (MIN)	QUANTITY (LB)	SIZE	FUNCTION
1.	FUEL TRANSFER	LH_2	10 MAX		270 MAX	30	3100 MAX	3"	Propellant Loading
1.1	FUEL DRAIN	LH_2							
2.	OXIDIZER TRANSFER	LOX	15 MAX		1620 MAX	30	4860 MAX	3"	Propellant Loading
2.1	OXIDIZER DRAIN	LOX						3"	
3.	THERMO-DYNAMIC VENT TO GSE VAC.	GOX							
4.	GOX VENT (RH)	O_2	15-16 23-24	660 163	8.10 MAX	N/A	N/A	3"	Relief
5.	GH_2 VENT	H_2	16-17 23-24	660 37	11.10 MAX	N/A	N/A	2"	Relief
6.	THERMO-DYNAMIC VENT TO GSE VAC.	H_2							
7.	GOX ACCUM. FILL	GOX, HE	1300 MAX	400	30 MAX	1.0	15	1/2"	He for purge and C/O ambient temperature
8.	GH_2 ACCUM. FILL	H_2, HE	1300 MAX	200	6 MAX	1.0	3	1/2"	He for purge and C/O ambient temperature
9.	HE BOTTLE FILL	HE	3500 MAX	AMB	8-10	15	92	1/2"	Stage safing

- Time Required

  The approximate total time required to perform the functional operation.

- Quantity

  In purging operations, it is the total quantity of fluid to be transferred across the interface. In loading operations, it is the total quantity of fluid required on board the Space Tug required for liftoff.

- Disconnect Requirements

  The number, size and location of GSE interface disconnect(s) necessary for performing the functional operations.

5.0 PHYSICAL REQUIREMENTS

The Space Tug/~~GSE physical interfaces shall be in~~ accordance with Figure TBD*(similar to V7-941514 and V7-941510).

*Figures 2.11-1, 2.11-3, 2.11-4 & 2.0-1 of Volume III.

Space Division
North American Rockwell

VOL II

APPENDIX D

DETAILED EXPENDABLES SCHEDULE

APPENDIX D

DETAILED EXPENDABLES SCHEDULE

The flight events timeline and consumables history presented in Table D-1 were obtained by means of an NR developed computer program; "Automated Mission-Timeline Analysis of Space Vehicles (Computer Aided Design IR&D)," modified for application to the Tug study. Criteria used in the development of the tabulated data and certain characteristics of the parameters not described in the expendable summary (Section 2.3 and Figure 2.3-1) are discussed below.

Table D-1 is comprised of three parts. The first part presents an assessment of required flight operations. The sequence and duration of events are based on a preliminary analysis. Part 2 presents the weight history of the Tug and delta-v's gained during propellant settling APS, main engine childown idle mode and main engine rated thrust burns. The main engine, APS, and fuel cell reactant propellants are all supplied from the LOX and LH_2 tanks. However, in order to enhance visibility, these propellants were allotted as separate initial starting weights. The consumption rate from the fuel cell reactant supply is comprised of the sum of a fuel cell consumption rate of 0.824 lb/hr and a conditioning for accumulator recharge rate of 0.12 lb/hr.

The "inert" weight listed at the end of Part 2 of 5338 pounds consists of the empty Tug weight of 3938 pounds and 1400 pounds of Shuttle Orbiter-to-Tug docking structure and safing equipment/He gas. During the tank safing process, 23.5 pounds of He is expended. Also, the total value of main engine delta-v (main DV) listed at the end of Part 2 includes allowances for gravity and steering losses, and a 2 percent contingency allowance for each burn. Since the Shuttle Orbiter was assumed to be the active vehicle during docking with the Tug, the back-up delta-v of 25 feet/sec (for use if the Tug needs to be the active vehicle) was not used and is not included as part of "Main DV." Prior to propellant dump and after mission completion, the residual propellant (useable reserve plus trapped plus gas) is indicated to be 1788 pounds.

Part 3 of Table D-1 presents a history of propellants expended during main engine and APS burns and propellant required for thermal conditioning of the feedlines (column heading is "Thermal Prop."). This conditioning propellant was assumed to be provided by the APS supply.

The equations programmed to calculate the attitude control functions ("attitude hold," "attitude maneuver," "stab control," "roll control"), moment of inertias ("roll moment," "yaw moment" assumed equal to pitch moment) and "translation prop." (using the APS) are derived in SD71-292-5, "Pre-Phase A Study for Analysis of a Reusable Space Tug." Pitch, yaw and roll rates which occur during main engine shutdown were assumed to be 0.5 deg/sec and were used as the initial conditions for the calculations of "Stab Control."

The expression "roll control" refers to control of roll rate buildup with the ACS during main engine burn of the Tug. A deadband of 5 degrees and a reaction jet pulse width of 50 milliseconds was used for the limit cycling attitude hold calculations, except when a dead band of 1 degree was used during docking with the Orbiter.

The moment of inertia calculations were suppressed while the Tug was stowed in the Orbiter. A total of 28 attitude maneuvers were programmed. In order to allow capability for a spherical change in attitude, each maneuver was performed by pitching or yawing 180 degrees in 200 seconds and rolling 90 degrees in 100 seconds. An attitude maneuver was programmed prior to each IMU alignment, to orient the Tug prior to each main engine or APS delta-v burn, during Tug separation from the Orbiter and during retrieval of the payload in synchronous orbit.

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 1 of 16)

MISSION EVENT NO	EVENT START TIME HR	MN	SC	EVENT DURATION TIME HR	MN	SC	FUNCTION BLOCK DSGNTN	MISSION PHASES AND EVENTS
								LAUNCH PHASE
1	0	0	0	0	0	6	1.1	LIFTOFF
2	0	0	6	0	0	2	1.1	PERFORM ROLL MANEUVER
3	0	0	8	0	0	16	1.1	PERFORM PITCH MANEUVER
4	0	0	24	0	0	41	1.1	PERFORM GRAVITY TURN MANEUVER
5	0	1	5	0	1	12	1.1	MAXIMUM DYNAMIC PRESSURE
6	0	2	17	0	1	1	1.1	MAXIMUM AXIAL LOAD (3G)
7	0	3	18	0	3	49	1.1	ORBITER ENGINES IGNITION
8	0	7	7	0	0	0	1.1	ORB MAIN ENGINE SHUTDOWN
9	0	7	7	0	0	5	1.1	ORBITER-BOOSTER STAGING
								OPERATIONS ORBIT
10	0	7	12	0	0	0	1.3	EARTH ORBIT ACHIEVED (50X100 NM)
11	0	7	12	0	43	48	1.3	COAST TO APOGEE
12	0	51	0	0	0	0	1.4.3	CIRCULARIZE ORBIT (100 NM)
13	0	51	0	0	10	0	1.4.4	COAST IN OPERATIONS ORBIT
14	1	1	0	0	2	0	1.4.7	ACTIVATE TUG SYSTEMS
15	1	3	0	0	23	0	1.4.7	VERIFY TUG SYSTEMS OPERATION INCLUDING INITIALIZE TUG STATE VECTOR WITH ORB
16	1	26	0	0	20	0	1.4.8	DEPLOY TUG
17	1	46	0	0	5	0	1.4.9	SEPARATE TUG FROM ORBITER
18	1	51	0	0	5	0	1.7.1	MANEUVER FOR IMU ALIGNMENT
19	1	56	0	0	3	0	1.7.1	PERFORM IMU ALIGNMENT
20	1	59	0	0	2	0	1.7.2	ORB VERIFY TUG STATE VECTOR
21	2	1	0	11	47	30	1.7	PHASING FOR AEI
22	13	48	30	0	5	0	1.7.1	MANEUVER FOR IMU ALIGNMENT
23	13	53	30	0	3	0	1.7.1	PERFORM IMU ALIGNMENT
24	13	56	30	0	1	30	1.7.4	COMPUTE DV PARAMETERS FOR AEI BURN
25	13	58	0	0	5	0	1.7.5	MANEUVER TO AEI BURN ATTITUDE
26	14	3	0	0	2	0	1.7.7	DOWNLINK SUBSYSTEMS STATUS
27	14	5	0	0	2	0	1.7.6	GO/NO-GO FOR AEI

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 2 of 16)

MISSION EVENT NO	EVENT START TIME HR	MN	SC	EVENT DURATION TIME HR	MN	SC	FUNCTION BLOCK DSGNTN	MISSION PHASES AND EVENTS
28	14	7	0	0	0	16	1.8	PROPELLANT SETTLING ACPS BURN
29	14	7	16	0	2	4	1.8.1	CHILLDOWN IDLE MODE MAIN ENG BURN
30	14	9	20	0	21	35	1.8.1	AEI BURN (100 X 19,300 NM)
								ASCENT TO APOGEE
31	14	30	55	0	2	0	1.8.4	DOWNLINK DV AND SUBSYSTEMS STATUS
32	14	32	55	1	30	40	1.8.5	VERIFY TUG POSITION,COURSE,VELOCITY
33	16	3	35	0	5	0	1.8.5	MANEUVER FOR IMU ALIGNMENT
34	16	8	35	0	3	0	1.8.5	PERFORM IMU ALIGNMENT
35	16	11	35	0	7	0	1.8.5	COMPUTE DV PARAMETERS FOR MCC BURN
36	16	18	35	0	5	0	1.8.5	MANEUVER TO MCC BURN ATTITUDE
37	16	23	35	0	2	0	1.8.6	DOWNLINK SUBSYSTEMS STATUS
38	16	25	35	0	2	0	1.8.6	GO/NO-GO FOR MCC
39	16	27	35	0	3	19	1.8.7	ACPS MCC BURN
40	16	30	54	0	2	0	1.8.9	DOWNLINK DV AND SUBSYSTEMS STATUS
41	16	32	54	2	43	0	1.8.8	VERIFY TUG POSITION,COURSE,VELOCITY
42	19	15	54	0	5	0	1.9.1	MANEUVER FOR IMU ALIGNMENT
43	19	20	54	0	3	0	1.9.1	PERFORM IMU ALIGNMENT
44	19	23	54	0	5	0	1.9.3	COMPUTE DV PARAMETERS FOR SOI
45	19	28	54	0	5	0	1.9.4	MANEUVER TO SOI BURN ATTITUDE
46	19	33	54	0	2	0	1.9.6	DOWNLINK SUBSYSTEMS STATUS
47	19	35	54	0	2	0	1.9.6	GO/NO-GO FOR SOI
48	19	37	54	0	0	12	1.9	PROPELLANT SETTLING ACPS BURN
49	19	38	6	0	2	4	1.9.7	CHILLDOWN IDLE MODE MAIN ENG BURN
50	19	40	10	0	9	10	1.9.7	SOI BURN
								SYNCHRONOUS ORBIT
51	19	49	20	0	2	0	1.10.4	DOWNLINK DV AND SUBSYSTEMS STATUS
52	19	51	20	0	2	0	1.10.1	UPDATE STATE VECTOR
53	19	53	20	0	5	0	1.10.2	DETERMINE VARIANCE FROM PLANNED ORBIT
54	19	58	20	0	5	0	1.10.3	COMPUTE PARAMETERS TO CORRECT ORBIT

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 3 of 16)

MISSION EVENT NO	EVENT START TIME HR	MN	SC	EVENT DURATION TIME HR	MN	SC	FUNCTION BLOCK DSGNTN	MISSION PHASES AND EVENTS
55	20	3	20	0	1	20	1.10.5	PERFORM ACPS BURN
56	20	4	40	0	2	0	1.10.6	REPORT STATUS TO MISSION CONTROL
57	20	6	40	0	5	0	1.10.7	ORIENT TUG TO PAYLOAD DEPLOYMENT ATT
58	20	11	40	0	5	0	1.10.8	SEPARATE TUG FROM PAYLOAD
59	20	16	40	0	3	0	1.10.9	VERIFY ADEQUATE SEPARATION DISTANCE
60	20	19	40	0	9	0	1.10.10	VERIFY PAYLOAD OPERATION
61	20	28	40	0	3	0	1.10.11	STOW TUG DEPLOYMENT MECHANISMS
62	20	31	40	0	2	0	1.10.14	DOWNLINK SUBSYSTEMS STATUS
63	20	33	40	0	5	0	1.19.1	MANEUVER FOR IMU ALIGNMENT
64	20	38	40	0	3	0	1.19.1	PERFORM IMU ALIGNMENT
65	20	41	40	0	5	0	1.19.3	COMPUTE DV PARAMETERS FOR POI
66	20	46	40	0	5	0	1.19.4	MANEUVER TO PHASING ORBIT BURN ATT
67	20	51	40	0	3	52	1.19.7	PHASING ORBIT INSERTION ACPS BURN
								PAYLOAD RETRIEVAL PHASING ORBIT
68	20	55	32	0	2	0	1.19.8	DOWNLINK DV AND SUBSYSTEMS STATUS
69	20	57	32	73	40	0	1.19.8	PHASING FOR PAYLOAD RETRIEVAL
70	94	37	32	0	5	0	1.19.8	MANEUVER FOR IMU ALIGNMENT
71	94	42	32	0	3	0	1.19.8	PERFORM IMU ALIGNMENT
72	94	45	32	0	7	0	1.19.8	COMPUTE DV PARAMETERS FOR RENDEZ BURNS
73	94	52	32	0	5	0	1.19.8	MANEUVER TO RENDEZVOUS BURN ATT
74	94	57	32	0	3	48	1.19.9	RENDEZVOUS ACPS BURNS
								RETRIEVE PAYLOAD AND COAST IN SYNC ORB
75	95	1	20	0	2	0	1.20.7	DOWNLINK DV AND SUBSYSTEMS STATUS
76	95	3	20	0	5	0	1.20.1	ACQUIRE AND LOCK ON TO PAYLOAD
77	95	8	20	0	5	0	1.20	MANEUVER FOR IMU ALIGNMENT
78	95	13	20	0	3	0	1.20	PERFORM IMU ALIGNMENT
79	95	16	20	0	10	0	1.20.18	READY TUG DOCKING MECHANISM
80	95	26	20	0	5	0	1.20.17	MANEUVER TO DOCKING ATTITUDE
81	95	31	20	0	10	0	1.20.16	TRANSLATE AND MANEUVER FOR DOCKING

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 4 of 16)

MISSION EVENT NO	EVENT START TIME HR	MN	SC	EVENT DURATION TIME HR	MN	SC	FUNCTION BLOCK DSGNTN	MISSION PHASES AND EVENTS
82	95	41	20	0	20	0	1.20.19	DOCK AND ATTACH PAYLOAD
83	96	1	20	0	30	0		SAFE PAYLOAD
84	96	31	20	11	28	0	1.11	PHASING FOR DEI
85	107	59	20	0	2	0	1.11.5	DOWNLINK DV AND SUBSYSTEMS STATUS
86	108	1	20	0	5	0	1.11.1	MANEUVER FOR IMU ALIGNMENT
87	108	6	20	0	3	0	1.11.1	PERFORM IMU ALIGNMENT
88	108	9	20	0	5	0	1.11.2	COMPUTE DV PARAMETERS FOR DEI BURN
89	108	14	20	0	5	0	1.11.3	MANEUVER TO DEI BURN ATTITUDE
90	108	19	20	0	2	0	1.11.5	DOWNLINK SUBSYSTEM STATUS
91	108	21	20	0	2	0	1.11.5	GO/NO-GO FOR DEI
92	108	23	20	0	0	10	1.11	PROPELLANT SETTLING ACPS BURN
93	108	23	30	0	7	0	1.11	VENT PRIOR TO DEI BURN
94	108	30	30	0	2	16	1.11.6	CHILLDOWN IDLE MODE MAIN ENG BURN
95	108	32	46	0	5	55	1.11.6	DEI BURN (19,300 X 270 NM)
								DESCENT TO PHASING ORBIT (270 X 270)
96	108	38	41	0	2	0	1.11.7	DOWNLINK DV AND SUBSYSTEMS STATUS
97	108	40	41	1	45	0	1.11.7	VERIFY TUG POSITION,COURSE,VELOCITY
98	110	25	41	0	5	0	1.11.7	MANEUVER FOR IMU ALIGNMENT
99	110	30	41	0	3	0	1.11.7	PERFORM IMU ALIGNMENT
100	110	33	41	0	5	0	1.11.7	COMPUTE DV PARAMETERS FOR MCC
101	110	38	41	0	5	0	1.11.7	MANEUVER TO MCC BURN ATTITUDE
102	110	43	41	0	2	0	1.11.7	DOWNLINK SUBSYSTEMS STATUS
103	110	45	41	0	2	0	1.11.7	GO/NO-GO FOR MCC
104	110	47	41	0	1	27	1.11.8	ACPS MCC BURN
105	110	49	8	0	2	0	1.11.10	DOWNLINK DV AND SUBSYSTEMS STATUS
106	110	51	8	2	57	0	1.11.10	VERIFY TUG POSITION COURSE,VELOCITY
107	113	48	8	0	5	0	1.12.1	MANEUVER FOR IMU ALIGNMENT
108	113	53	8	0	3	0	1.12.1	PERFORM IMU ALIGNMENT
109	113	56	8	0	5	0	1.12.3	COMPUTE DV PARAMETERS FOR POI
110	114	1	8	0	5	0	1.12.4	MANEUVER TO POI BURN ATT
111	114	6	8	0	2	0	1.12.6	DOWNLINK SUBSYSTEMS STATUS

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval (Sheet 5 of 16)

MISSION EVENT NO	EVENT START TIME HR	MN	SC	EVENT DURATION TIME HR	MN	SC	FUNCTION BLOCK DSGNTN	MISSION PHASES AND EVENTS
112	114	8	8	0	0	8	1.12	PROPELLANT SETTLING ACPS BURN
113	114	8	16	0	1	59	1.12.7	CHILLDOWN IDLE MODE MAIN ENG BURN
114	114	10	15	0	5	2	1.12.7	POI BURN
								PHASING ORBIT (270 X 270)
115	114	15	17	0	2	0	1.12.10	DOWNLINK DV AND SUBSYSTEMS STATUS
116	114	17	17	21	47	30	1.12.11	PHASING FOR OOTI
117	136	4	47	0	5	0	1.13.1	MANEUVER FOR IMU ALIGNMENT
118	136	9	47	0	3	0	1.13.1	PERFORM IMU ALIGNMENT
119	136	12	47	0	5	0	1.13.3	COMPUTE DV PARAMETERS FOR OOTI
120	136	17	47	0	5	0	1.13.4	MANEUVER TO OOTI BURN ATT
121	136	22	47	0	0	6	1.13	PROPELLANT SETTLING ACPS BURN
122	136	22	53	0	1	58	1.13.7	CHILLDOWN IDLE MODE MAIN ENG BURN
123	136	24	51	0	0	8	1.13.7	OOTI BURN
								RENDEZVOUS WITH ORBITER
124	136	24	59	0	2	0	1.13.8	DOWNLINK DV AND SUBSYSTEMS STATUS
125	136	26	59	0	33	0	1.13.8	VERIFY TUG POSITION,COURSE,VELOCITY
126	136	59	59	0	5	0	1.13.8	MANEUVER FOR IMU ALIGNMENT
127	137	4	59	0	3	0	1.13.8	PERFORM IMU ALIGNMENT
128	137	7	59	0	5	0	1.13.8	COMPUTE DV PARAMETERS FOR OOI
129	137	12	59	0	5	0	1.13.8	MANEUVER TO OOI BURN ATTITUDE
130	137	17	59	0	0	6	1.13	PROPELLANT SETTLING ACPS BURN
131	137	18	5	0	1	59	1.13.9	CHILLDOWN IDLE MODE MAIN ENG BURN
132	137	20	4	0	0	8	1.13.9	OOI BURN
133	137	20	12	0	2	0	1.13.10	DOWNLINK STATUS REPORT
134	137	22	12	0	46	0		TUG FORWARD OF AND ABOVE ORB
135	138	8	12	0	5	0	1.14	MANEUVER FOR IMU ALIGNMENT
136	138	13	12	0	3	0	1.14	PERFORM IMU ALIGNMENT
137	138	16	12	0	5	0	1.14	COMPUTE TPI DV PARAMETERS
138	138	21	12	0	5	0	1.14	MANEUVER TO TPI ATTITUDE

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 6 of 16)

MISSION EVENT NO	EVENT START TIME HR	MN	SC	EVENT DURATION TIME HR	MN	SC	FUNCTION BLOCK DSGNTN	MISSION PHASES AND EVENTS
139	138	26	12	0	0	49	1.14.4	TUG TPI ACPS BURNS
140	138	27	1	1	0	0	1.14.5	TUG PREPARATIONS FOR TPF BURNS
141	139	27	1	0	0	48	1.14.5	TUG TPF ACPS BURNS
								TUG MAINTAIN ATTITUDE HOLD
142	139	27	49	0	3	0	1.14.1	SIGNAL ACQ AND LOCK-ON BY ORBITER
143	139	30	49	0	3	0	1.14.2	EST. COMM. BETWEEN VEHICLES
144	139	33	49	0	27	0	1.14.12	CONFIG. FOR DOCKING
145	140	0	49	0	27	0	1.14.13	CLOSURE BY ORBITER AND DOCK
								OPERATIONS ORBIT
146	140	27	49	0	20	0	1.15.1	RETRACT TUG AND STOW IN ORBITER
147	140	47	49	0	1	15	1.15.3	DUMP AND PURGE PROPELLANTS
148	140	49	4	0	3	11	1.15.3	PRESSURIZE TANKS TO 17 PSIA
149	140	52	15	0	2	19	1.15.3	BLOWDOWN TO 1 PSIA
150	140	54	34	0	3	45	1.15.3	PRESSURIZE TANKS TO 17 PSIA
151	140	58	19	0	2	30	1.15.4	COMPLETE TUG DEACTIVATION
152	141	0	49	15	0	0	1.15.4	ORBITER PHASE TO RETURN TO EARTH
153	156	0	49	0	8	0	1.16	DEORBIT
154	156	8	49	0	34	0	1.16	REENTRY
155	156	42	49	0	0	0	1.16	LAND

Table D-1 Tug Geosynchronous Base ine Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 7 of 16)

MISSION EVENT NO.	MISSION EVENT TIME HR	MN	SC	DELTA-V REQD FT/SEC	PAYLOAD WT LBS	MAIN PROPELLANT SUPPLY LBS	ACPS PROPELLANT SUPPLY LBS	FUEL CELL REACTANT SUPPLY LBS	TOTAL VEHICLE WT LBS	NET CHANGE LBS
1	0	0	0	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
2	0	0	6	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
3	0	0	8	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
4	0	0	24	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
5	0	1	5	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
6	0	2	17	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
7	0	3	18	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
8	0	7	7	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
9	0	7	7	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
10	0	7	12	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
11	0	7	12	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
12	0	51	0	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
13	0	51	0	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
14	1	1	0	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
15	1	3	0	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.0
16	1	26	0	0.0	3000.0	55302.0	1200.0	160.0	65000.0	0.5
17	1	46	0	10.0	3000.0	55302.0	1199.8	159.6	63599.4	1400.4
18	1	51	0	0.0	3000.0	55302.0	1199.7	159.3	63599.0	54.7
19	1	56	0	0.0	3000.0	55302.0	1145.1	159.2	63544.3	2.2
20	1	59	0	0.0	3000.0	55302.0	1142.9	159.1	63542.1	0.1
21	2	1	0	0.0	3000.0	55302.0	1142.9	159.1	63542.0	0.1
22	13	48	30	0.0	3000.0	55302.0	1142.9	159.1	63541.9	18.3
23	13	53	30	0.0	3000.0	55302.0	1135.7	147.9	63523.7	2.2
24	13	56	30	0.0	3000.0	55302.0	1133.6	147.8	63521.4	0.1
25	13	58	0	0.0	3000.0	55302.0	1133.6	147.8	63521.4	0.0
26	14	3	0	0.0	3000.0	55302.0	1133.5	147.8	63521.3	2.2
27	14	5	0	0.0	3000.0	55302.0	1131.4	147.7	63519.1	0.1
28	14	7	0	2.2	3000.0	55302.0	1131.4	147.7	63519.0	0.1
29	14	7	16	2.1	3000.0	55302.0	1131.4	147.6	63519.0	11.9
30	14	9	20	8610.7	3000.0	55302.0	1119.4	147.6	63507.1	11.9
31	14	30	55	0.0	3000.0	55291.3	1118.3	147.6	63495.2	27568.3
32	14	32	55	0.0	3000.0	27724.7	1116.9	147.3	35926.8	0.1
33	16	3	35	0.0	3000.0	27724.7	1116.9	147.2	35926.8	2.4
						27724.7	1116.0	145.8	35924.4	2.1

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 8 of 16)

MISSION EVENT NO.	MISSION EVENT TIME HR MN SC	DELTA-V REQD FT/SEC	PAYLOAD WT LBS	MAIN PROPELLANT SUPPLY LBS	ACPS PROPELLANT SUPPLY LBS	FUEL CELL REACTANT SUPPLY LBS	TOTAL VEHICLE WT LBS	NET CHANGE LBS
34	16 8 35	0.0	3000.0	27724.7	1113.9	145.7	35922.3	0.1
35	16 11 35	0.0	3000.0	27724.7	1113.9	145.7	35922.3	0.2
36	16 18 35	0.0	3000.0	27724.7	1113.8	145.6	35922.1	2.1
37	16 23 35	0.0	3000.0	27724.7	1111.8	145.5	35920.0	0.1
38	16 25 35	0.0	3000.0	27724.7	1111.8	145.5	35919.9	0.1
39	16 27 35	50.0	3000.0	27724.7	1111.8	145.4	35919.9	146.6
40	16 30 54	0.0	3000.0	27724.7	965.3	145.4	35773.3	0.1
41	16 32 54	0.0	3000.0	27724.7	965.2	145.3	35773.2	4.2
42	19 15 54	0.0	3000.0	27724.7	963.6	142.8	35769.0	2.1
43	19 20 54	0.0	3000.0	27724.7	961.5	142.7	35766.9	0.1
44	19 23 54	0.0	3000.0	27724.7	961.5	142.6	35766.8	0.1
45	19 28 54	0.0	3000.0	27724.7	961.5	142.6	35766.7	2.1
46	19 33 54	0.0	3000.0	27724.7	959.4	142.5	35764.6	0.1
47	19 35 54	0.0	3000.0	27724.7	959.4	142.5	35764.5	0.1
48	19 37 54	2.9	3000.0	27724.7	959.4	142.4	35764.5	8.7
49	19 38 6	3.8	3000.0	27724.7	950.7	142.4	35755.8	11.8
50	19 40 10	6003.3	3000.0	27714.0	949.6	142.4	35744.0	11713.5
51	19 49 20	0.0	3000.0	16001.8	948.4	142.2	24030.5	0.1
52	19 51 20	0.0	3000.0	16001.8	948.4	142.2	24030.5	0.1
53	19 53 20	0.0	3000.0	16001.8	948.4	142.2	24030.4	0.1
54	19 58 20	0.0	3000.0	16001.8	948.3	142.1	24030.3	0.1
55	20 3 20	30.0	3000.0	16001.8	948.3	142.0	24030.[illegible]	58.9
56	20 4 40	0.0	3000.0	16001.8	889.4	142.0	23971.[illegible]	0.1
57	20 6 40	0.0	3000.0	16001.8	889.3	142.0	23971.[illegible]	2.0
58	20 11 40	10.0	0.0	16001.8	887.4	141.9	20969.2	3018.2
59	20 16 40	0.0	0.0	16001.8	869.4	141.8	20951.0	0.1
60	20 19 40	0.0	0.0	16001.8	869.3	141.8	20950.9	0.3
61	20 28 40	0.0	0.0	16001.8	869.2	141.6	20950.6	0.1
62	20 31 40	0.0	0.0	16001.8	869.1	141.6	20950.5	0.1
63	20 33 40	0.0	0.0	16001.8	869.1	141.5	20950.4	0.6
64	20 38 40	0.0	0.0	16001.8	868.5	141.5	20949.8	0.1
65	20 41 40	0.0	0.0	16001.8	868.5	141.4	20949.7	0.2
66	20 46 40	0.0	0.0	16001.8	868.4	141.3	20949.5	0.6

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 9 of 16)

MISSION EVENT NO.	MISSION EVENT TIME HR MN SC	DELTA-V REQD FT/SEC	PAYLOAD WT LBS	MAIN PROPELLANT SUPPLY LBS	ACPS PROPELLANT SUPPLY LBS	FUEL CELL REACTANT SUPPLY LBS	TOTAL VEHICLE WT LBS	NET CHANGE LBS
67	20 51 40	100.0	0.0	16001.8	867.8	141.3	20948.9	170.5
68	20 55 32	0.0	0.0	16001.8	697.4	141.2	20778.4	0.1
69	20 57 32	0.0	0.0	16001.8	697.4	141.2	20778.4	145.4
70	94 37 32	0.0	0.0	16001.8	621.5	71.6	20632.9	0.6
71	94 42 32	0.0	0.0	16001.8	620.9	71.6	20632.3	0.1
72	94 45 32	0.0	0.0	16001.8	620.9	71.5	20632.2	0.2
73	94 52 32	0.0	0.0	16001.8	620.7	71.4	20632.0	0.6
74	94 57 32	100.0	0.0	16001.8	620.2	71.3	20631.3	167.9
75	95 1 20	0.0	0.0	16001.8	452.4	71.3	20463.4	0.1
76	95 3 20	0.0	0.0	16001.8	452.3	71.2	20463.4	0.2
77	95 8 20	0.0	0.0	16001.8	452.2	71.1	20463.2	0.6
78	95 13 20	0.0	0.0	16001.8	451.7	71.1	20462.6	0.1
79	95 16 20	0.0	0.0	16001.8	451.6	71.0	20462.5	0.3
80	95 26 20	0.0	0.0	16001.8	451.5	70.9	20462.1	0.6
81	95 31 20	15.0	0.0	16001.8	450.9	70.8	20461.5	26.4
82	95 41 20	0.0	3000.0	16001.8	424.6	70.6	23435.1	-2999.5
83	96 1 20	0.0	3000.0	16001.8	424.4	70.3	23434.5	0.8
84	96 31 20	0.0	3000.0	16001.8	424.1	69.8	23433.8	17.9
85	107 59 20	0.0	3000.0	16001.8	417.0	59.0	23415.8	0.1
86	108 1 20	0.0	3000.0	16001.8	416.9	59.0	23415.8	2.0
87	108 6 20	0.0	3000.0	16001.8	415.1	58.9	23413.8	0.1
88	108 9 20	0.0	3000.0	16001.8	415.0	58.9	23413.7	0.1
89	108 14 20	0.0	3000.0	16001.8	415.0	58.8	23413.6	2.0
90	108 19 20	0.0	3000.0	16001.8	413.1	58.7	23411.6	0.1
91	108 21 20	0.0	3000.0	16001.8	413.0	58.7	23411.5	0.1
92	108 23 20	3.9	3000.0	16001.8	413.0	58.6	23411.5	7.6
93	108 23 30	0.0	3000.0	16001.8	405.4	58.6	23403.9	83.2
94	108 30 30	6.5	3000.0	15918.8	405.3	58.5	23320.7	12.8
95	108 32 46	5926.6	3000.0	15907.1	404.3	58.5	23307.9	7558.6
96	108 38 41	0.0	3000.0	8349.5	403.3	58.4	15749.2	0.1
97	108 40 41	0.0	3000.0	8349.5	403.3	58.4	15749.2	2.8
98	110 25 41	0.0	3000.0	8349.5	402.2	56.7	15746.4	1.8
99	110 30 41	0.0	3000.0	8349.5	400.5	56.6	15744.6	0.1

Space Division
North American Rockwell

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 10 of 16)

MISSION EVENT NO.	MISSION EVENT TIME HR MN SC	DELTA-V REQD FT/SEC	PAYLOAD WT LBS	MAIN PROPELLANT SUPPLY LBS	ACPS PROPELLANT SUPPLY LBS	FUEL CELL REACTANT SUPPLY LBS	TOTAL VEHICLE WT LBS	NET CHANGE LBS
100	110 33 41	0.0	3000.0	8349.5	400.4	56.6	15744.5	0.1
101	110 38 41	0.0	3000.0	8349.5	400.4	56.5	15744.4	1.8
102	110 43 41	0.0	3000.0	8349.5	398.7	56.4	15742.6	0.1
103	110 45 41	0.0	3000.0	8349.5	398.6	56.4	15742.6	0.1
104	110 47 41	50.0	3000.0	8349.5	398.6	56.4	15742.5	64.2
105	110 49 8	0.0	3000.0	8349.5	334.4	56.3	15678.3	0.1
106	110 51 8	0.0	3000.0	8349.5	334.4	56.3	15678.2	4.7
107	113 48 8	0.0	3000.0	8349.5	332.5	53.5	15673.6	1.8
108	113 53 8	0.0	3000.0	8349.5	330.8	53.5	15671.8	0.1
109	113 56 8	0.0	3000.0	8349.5	330.8	53.4	15671.7	0.1
110	114 1 8	0.0	3000.0	8349.5	330.7	53.3	15671.6	1.8
111	114 6 8	0.0	3000.0	8349.5	329.0	53.2	15669.8	0.1
112	114 8 8	4.4	3000.0	8349.5	329.0	53.2	15669.7	5.7
113	114 8 16	8.5	3000.0	8349.5	323.3	53.2	15664.0	11.2
114	114 10 15	8012.1	3000.0	8339.2	322.3	53.2	15652.7	6438.9
115	114 15 17	0.0	3000.0	1901.2	321.6	53.1	9213.9	0.1
116	114 17 17	0.0	3000.0	1901.2	321.6	53.1	9213.8	35.4
117	136 4 47	0.0	3000.0	1901.2	306.7	32.5	9178.4	1.4
118	136 9 47	0.0	3000.0	1901.2	305.4	32.4	9177.0	0.1
119	136 12 47	0.0	3000.0	1901.2	305.3	32.4	9176.9	0.1
120	136 17 47	0.0	3000.0	1901.2	305.3	32.3	9176.8	1.4
121	136 22 47	5.7	3000.0	1901.2	304.0	32.2	9175.4	4.3
122	136 22 53	14.4	3000.0	1901.2	299.7	32.2	9171.1	10.9
123	136 24 51	281.9	3000.0	1891.1	299.0	32.2	9160.2	169.9
124	136 24 59	0.0	3000.0	1721.8	298.3	32.2	8990.3	0.1
125	136 26 59	0.0	3000.0	1721.8	298.3	32.1	8990.2	0.9
126	136 59 59	0.0	3000.0	1721.8	297.9	31.6	8989.3	1.4
127	137 4 59	0.0	3000.0	1721.8	296.6	31.6	8988.0	0.1
128	137 7 59	0.0	3000.0	1721.8	296.6	31.5	8987.9	0.1
129	137 12 59	0.0	3000.0	1721.8	296.5	31.4	8987.8	1.4
130	137 17 59	5.8	3000.0	1721.8	295.2	31.3	8986.4	4.3
131	137 18 5	14.8	3000.0	1721.8	290.9	31.3	8982.1	10.9
132	137 20 4	281.4	3000.0	1711.6	290.2	31.3	8971.2	166.1

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 11 of 16)

MISSION EVENT NO.	MISSION EVENT TIME HR MN SC	DELTA-V REQD FT/SEC	PAYLOAD WT LBS	MAIN PROPELLANT SUPPLY LBS	ACPS PROPELLANT SUPPLY LBS	FUEL CELL REACTANT SUPPLY LBS	TOTAL VEHICLE WT LBS	NET CHANGE LBS
133	137 20 12	0.0	3000.0	1546.1	289.6	31.3	8805.0	0.1
134	137 22 12	0.0	3000.0	1546.1	289.6	31.3	8805.0	1.3
135	138 8 12	0.0	3000.0	1546.1	289.0	30.6	8803.7	1.3
136	138 13 12	0.0	3000.0	1546.1	287.9	30.5	8802.4	0.1
137	138 16 12	0.0	3000.0	1546.1	287.7	30.4	8802.3	0.1
138	138 21 12	0.0	3000.0	1546.1	287.7	30.4	8802.2	1.3
139	138 26 12	50.0	3000.0	1546.1	286.4	30.3	8800.8	35.9
140	138 27 1	0.0	3000.0	1546.1	250.5	30.3	8764.9	1.6
141	139 27 1	50.0	3000.0	1546.1	249.8	29.3	8763.3	35.8
142	139 27 49	0.0	3000.0	1546.1	214.1	29.3	8727.5	0.1
143	139 30 49	0.0	3000.0	1546.1	214.1	29.3	8727.5	0.1
144	139 33 49	0.0	3000.0	1546.1	214.0	29.2	8727.4	0.7
145	140 0 49	0.0	3000.0	1546.1	213.7	28.8	8726.6	0.6
146	140 27 49	0.0	3000.0	1546.1	213.5	28.4	10126.0	-1399.6
147	140 47 49	0.0	3000.0	91.0	0.0	0.0	8429.0	1697.0
148	140 49 4	0.0	3000.0	91.0	0.0	0.0	8429.0	0.0
149	140 52 15	0.0	3000.0	13.5	0.0	0.0	8328.0	101.0
150	140 54 34	0.0	3000.0	13.5	0.0	0.0	8328.0	0.0
151	140 58 19	0.0	3000.0	13.5	0.0	0.0	8328.0	0.0
152	141 0 49	0.0	3000.0	13.5	0.0	0.0	8328.0	0.0
153	156 0 49	0.0	3000.0	13.5	0.0	0.0	8328.0	0.0
154	156 8 49	0.0	3000.0	13.5	0.0	0.0	8328.0	0.0
155	156 42 49	0.0	3000.0	13.5	0.0	0.0	8328.0	0.0

MAIN DV 29166.1 INERT WT 5338.0
ACS DV 489.9

TOTAL DV 29656.0

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 12 of 16)

MISSION SEGMENT NO.	MISSION SEGMENT DURATION HR	MAIN ENG THRUST PROP. LBS	VENT PROP. LBS	TRANS-LATION PROP. LBS	ATTITUDE HOLD PROP. LBS	ATTITUDE MANEUVER PROP. LBS	STAB. CONTROL PROP. LBS	ROLL CONTROL PROP. LBS	THERMAL PROP. LBS	ROLL MOMENT SLUG-FT**2	YAW MOMENT SLUG-FT**2
1 -2	0.003	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
2 -3	0.001	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
3 -4	0.004	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
4 -5	0.011	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
5 -6	0.020	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
6 -7	0.017	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
7 -8	0.064	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
8 -9	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
9 -10	0.001	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
10 -11	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
11 -12	0.730	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
12 -13	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
13 -14	0.167	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
14 -15	0.033	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
15 -16	0.383	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.2	0.	0.
16 -17	0.333	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.1	0.	0.
17 -18	0.083	0.0	0.0	52.4	0.086	2.119	0.0	0.0	0.0	7607.	163775.
18 -19	0.083	0.0	0.0	0.0	0.0	2.119	0.0	0.0	0.0	7607.	163775.
19 -20	0.050	0.0	0.0	0.0	0.010	0.0	0.0	0.0	0.0	7607.	163757.
20 -21	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	163757.
21 -22	11.792	0.0	0.0	0.0	2.422	0.0	0.0	0.0	4.7	7607.	163757.
22 -23	0.083	0.0	0.0	0.0	0.0	2.119	0.0	0.0	0.0	7607.	163757.
23 -24	0.050	0.0	0.0	0.0	0.010	0.0	0.0	0.0	0.0	7607.	163757.
24 -25	0.025	0.0	0.0	0.0	0.005	0.0	0.0	0.0	0.0	7607.	163749.
25 -26	0.083	0.0	0.0	0.0	0.0	2.119	0.0	0.0	0.0	7607.	163750.
26 -27	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	163749.
27 -28	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	163749.
28 -29	0.005	0.0	0.0	11.9	0.001	0.0	0.0	0.0	0.0	7607.	163749.
29 -30	0.035	10.7	0.0	0.0	0.0	0.0	1.137	0.015	0.0	7607.	163749.
30 -31	0.360	27566.6	0.0	0.0	0.0	0.0	1.067	0.151	0.0	7607.	163749.
31 -32	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.1	7607.	163749.
32 -33	1.511	0.0	0.0	0.0	0.321	0.0	0.0	0.0	0.0	7607.	153480.
33 -34	0.083	0.0	0.0	0.0	0.0	1.990	0.0	0.0	0.6	7607.	153480.

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval (Sheet 13 of 16)

MISSION SEGMENT NO.	MISSION SEGMENT DURATION HR	MAIN ENG THRUST PROP. LBS	VENT PROP. LBS	TRANS-LATION PROP. LBS	ATTITUDE HOLD PROP. LBS	ATTITUDE MANEUVER PROP. LBS	STAB. CONTROL PROP. LBS	ROLL CONTROL PROP. LBS	THERMAL PROP. LBS	ROLL MOMENT SLUG-FT**2	YAW MOMENT SLUG-FT**2
34 -35	0.050	0.0	0.0	0.0	0.011	0.0	0.0	0.0	0.0	7607.	153477.
35 -36	0.117	0.0	0.0	0.0	0.025	0.0	0.0	0.0	0.0	7607.	153477.
36 -37	0.083	0.0	0.0	0.0	0.0	1.990	0.0	0.0	0.0	7607.	153477.
37 -38	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	153476.
38 -39	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	153476.
39 -40	0.055	0.0	0.0	146.5	0.012	0.0	0.0	0.0	0.0	7607.	153476.
40 -41	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	153399.
41 -42	2.717	0.0	0.0	0.0	0.578	0.0	0.0	0.0	1.1	7607.	153399.
42 -43	0.083	0.0	0.0	0.0	0.0	1.989	0.0	0.0	0.0	7607.	153399.
43 -44	0.050	0.0	0.0	0.0	0.011	0.0	0.0	0.0	0.0	7607.	153396.
44 -45	0.083	0.0	0.0	0.0	0.018	0.0	0.0	0.0	0.0	7607.	153395.
45 -46	0.083	0.0	0.0	0.0	0.0	1.989	0.0	0.0	0.0	7607.	153395.
46 -47	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	153394.
47 -48	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	153394.
48 -49	0.003	0.0	0.0	8.7	0.001	0.0	0.0	0.0	0.0	7607.	153394.
49 -50	0.034	10.6	0.0	0.0	0.0	0.0	1.067	0.014	0.0	7607.	153394.
50 -51	0.153	11712.2	0.0	0.0	0.0	0.0	1.005	0.064	0.1	7607.	153394.
51 -52	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	144258.
52 -53	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	144258.
53 -54	0.083	0.0	0.0	0.0	0.018	0.0	0.0	0.0	0.0	7607.	144258.
54 -55	0.083	0.0	0.0	0.0	0.018	0.0	0.0	0.0	0.0	7607.	144258.
55 -56	0.022	0.0	0.0	58.9	0.005	0.0	0.0	0.0	0.0	7607.	144258.
56 -57	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	144188.
57 -58	0.083	0.0	0.0	0.0	0.0	1.873	0.0	0.0	0.0	7607.	144188.
58 -59	0.083	0.0	0.0	17.3	0.262	0.528	0.0	0.0	0.0	4317.	39026.
59 -60	0.050	0.0	0.0	0.0	0.031	0.0	0.0	0.0	0.0	4317.	39011.
60 -61	0.150	0.0	0.0	0.0	0.094	0.0	0.0	0.0	0.1	4317.	39011.
61 -62	0.050	0.0	0.0	0.0	0.031	0.0	0.0	0.0	0.0	4317.	39011.
62 -63	0.033	0.0	0.0	0.0	0.021	0.0	0.0	0.0	0.0	4317.	39010.
63 -64	0.083	0.0	0.0	0.0	0.0	0.528	0.0	0.0	0.0	4317.	39010.
64 -65	0.050	0.0	0.0	0.0	0.031	0.0	0.0	0.0	0.0	4317.	39010.
65 -66	0.083	0.0	0.0	0.0	0.052	0.0	0.0	0.0	0.0	4317.	39010.
66 -67	0.083	0.0	0.0	0.0	0.0	0.528	0.0	0.0	0.0	4317.	39010.

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
(Sheet 14 of 16)

MISSION SEGMENT NO.	MISSION SEGMENT DURATION HR	MAIN ENG THRUST PROP. LBS	VENT PROP. LBS	TRANS-LATION PROP. LBS	ATTITUDE HOLD PROP. LBS	ATTITUDE MANEUVER PROP. LBS	STAB. CONTROL PROP. LBS	ROLL CONTROL PROP. LBS	THERMAL PROP. LBS	ROLL MOMENT SLUG-FT**2	YAW MOMENT SLUG-FT**2
67 -68	0.064	0.0	0.0	170.4	0.040	0.0	0.0	0.0	0.0	4317.	39009.
68 -69	0.033	0.0	0.0	0.0	0.021	0.0	0.0	0.0	0.0	4317.	38863.
69 -70	73.567	0.0	0.0	0.0	46.411	0.0	0.0	0.0	29.5	4317.	38863.
70 -71	0.083	0.0	0.0	0.0	0.0	0.525	0.0	0.0	0.0	4317.	38853.
71 -72	0.050	0.0	0.0	0.0	0.032	0.0	0.0	0.0	0.0	4317.	38739.
72 -73	0.117	0.0	0.0	0.0	0.074	0.0	0.0	0.0	0.0	4317.	38739.
73 -74	0.083	0.0	0.0	0.0	0.0	0.525	0.0	0.0	0.0	4317.	38739.
74 -75	0.063	0.0	0.0	167.8	0.040	0.0	0.0	0.0	0.0	4317.	38738.
75 -76	0.033	0.0	0.0	0.0	0.021	0.0	0.0	0.0	0.0	4317.	38594.
76 -77	0.083	0.0	0.0	0.0	0.053	0.0	0.0	0.0	0.0	4317.	38594.
77 -78	0.083	0.0	0.0	0.0	0.0	0.523	0.0	0.0	0.0	4317.	38594.
78 -79	0.050	0.0	0.0	0.0	0.032	0.0	0.0	0.0	0.0	4317.	38593.
79 -80	0.167	0.0	0.0	0.0	0.106	0.0	0.0	0.0	0.1	4317.	38593.
80 -81	0.083	0.0	0.0	0.0	0.0	0.523	0.0	0.0	0.0	4317.	38593.
81 -82	0.167	0.0	0.0	25.2	0.528	0.523	0.0	0.0	0.1	4317.	38592.
82 -83	0.333	0.0	0.0	0.0	0.074	0.0	0.0	0.0	0.1	7607.	143542.
83 -84	0.500	0.0	0.0	0.0	0.110	0.0	0.0	0.0	0.2	7607.	143541.
84 -85	11.467	0.0	0.0	0.0	2.532	0.0	0.0	0.0	4.6	7607.	143540.
85 -86	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	143518.
86 -87	0.033	0.0	0.0	0.0	0.0	1.865	0.0	0.0	0.0	7607.	143518.
87 -88	0.050	0.0	0.0	0.0	0.011	0.0	0.0	0.0	0.0	7607.	143516.
88 -89	0.083	0.0	0.0	0.0	0.018	0.0	0.0	0.0	0.0	7607.	143516.
89 -90	0.083	0.0	0.0	0.0	0.0	1.865	0.0	0.0	0.0	7607.	143516.
90 -91	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	143513.
91 -92	0.033	0.0	0.0	0.0	0.007	0.0	0.0	0.0	0.0	7607.	143513.
92 -93	0.003	0.0	0.0	7.6	0.001	0.0	0.0	0.0	0.0	7607.	143513.
93 -94	0.117	0.0	83.0	0.0	0.026	0.0	0.0	0.0	0.0	7607.	143504.
94 -95	0.038	11.7	0.0	0.0	0.0	0.0	0.999	0.016	0.0	7607.	143504.
95 -96	0.099	7557.6	0.0	0.0	0.0	0.0	0.901	0.041	0.0	7607.	143504.
96 -97	0.033	0.0	0.0	0.0	0.008	0.0	0.0	0.0	0.0	7607.	129025.
97 -98	1.750	0.0	0.0	0.0	0.411	0.0	0.0	0.0	0.7	7607.	129025.
98 -99	0.083	0.0	0.0	0.0	0.0	1.684	0.0	0.0	0.0	7607.	129025.
99-100	0.050	0.0	0.0	0.0	0.012	0.0	0.0	0.0	0.0	7607.	129012.

Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
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MISSION SEGMENT NO.	MISSION SEGMENT DURATION HR	MAIN ENG THRUST PROP. LBS	VENT PROP. LBS	TRANS-LATION PROP. LBS	ATTITUDE HOLD PROP. LBS	ATTITUDE MANEUVER PROP. LBS	STAB. CONTROL PROP. LBS	ROLL CONTROL PROP. LBS	THERMAL PROP. LBS	ROLL MOMENT SLUG-FT**2	YAW MOMENT SLUG-FT**2
100-101	0.083	0.0	0.0	0.0	0.020	0.0	0.0	0.0	0.0	7607.	129012.
101-102	0.083	0.0	0.0	0.0	0.0	1.684	0.0	0.0	0.0	7607.	129012.
102-103	0.033	0.0	0.0	0.0	0.008	0.0	0.0	0.0	0.0	7607.	129006.
103-104	0.033	0.0	0.0	0.0	0.008	0.0	0.0	0.0	0.0	7607.	129006.
104-105	0.024	0.0	0.0	64.2	0.006	0.0	0.0	0.0	0.0	7607.	129006.
105-106	0.033	0.0	0.0	0.0	0.008	0.0	0.0	0.0	0.0	7607.	128820.
106-107	2.950	0.0	0.0	0.0	0.693	0.0	0.0	0.0	1.2	7607.	128820.
107-108	0.083	0.0	0.0	0.0	0.0	1.681	0.0	0.0	0.0	7607.	128820.
108-109	0.050	0.0	0.0	0.0	0.012	0.0	0.0	0.0	0.0	7607.	128801.
109-110	0.083	0.0	0.0	0.0	0.020	0.0	0.0	0.0	0.0	7607.	128801.
110-111	0.083	0.0	0.0	0.0	0.0	1.681	0.0	0.0	0.0	7607.	128801.
111-112	0.033	0.0	0.0	0.0	0.008	0.0	0.0	0.0	0.0	7607.	128795.
112-113	0.002	0.0	0.0	5.7	0.001	0.0	0.0	0.0	0.0	7607.	128795.
113-114	0.033	10.3	0.0	0.0	0.0	0.0	0.900	0.014	0.0	7607.	128795.
114-115	0.084	6438.0	0.0	0.0	0.0	0.0	0.677	0.035	0.0	7607.	128795.
115-116	0.033	0.0	0.0	0.0	0.009	0.0	0.0	0.0	0.0	7607.	96017.
116-117	21.792	0.0	0.0	0.0	6.157	0.0	0.0	0.0	8.7	7607.	96017.
117-118	0.083	0.0	0.0	0.0	0.0	1.269	0.0	0.0	0.0	7607.	96017.
118-119	0.050	0.0	0.0	0.0	0.014	0.0	0.0	0.0	0.0	7607.	95692.
119-120	0.083	0.0	0.0	0.0	0.024	0.0	0.0	0.0	0.0	7607.	95691.
120-121	0.083	0.0	0.0	0.0	0.0	1.269	0.0	0.0	0.0	7607.	95691.
121-122	0.002	0.0	0.0	4.3	0.000	0.0	0.0	0.0	0.0	7607.	95678.
122-123	0.033	10.1	0.0	0.0	0.0	0.0	0.674	0.014	0.0	7607.	95678.
123-124	0.002	169.2	0.0	0.0	0.0	0.0	0.664	0.001	0.0	7607.	95678.
124-125	0.033	0.0	0.0	0.0	0.010	0.0	0.0	0.0	0.0	7607.	94003.
125-126	0.550	0.0	0.0	0.0	0.158	0.0	0.0	0.0	0.2	7607.	94002.
126-127	0.083	0.0	0.0	0.0	0.0	1.248	0.0	0.0	0.0	7607.	94002.
127-128	0.050	0.0	0.0	0.0	0.014	0.0	0.0	0.0	0.0	7607.	93981.
128-129	0.083	0.0	0.0	0.0	0.024	0.0	0.0	0.0	0.0	7607.	93980.
129-130	0.083	0.0	0.0	0.0	0.0	1.248	0.0	0.0	0.0	7607.	93980.
130-131	0.002	0.0	0.0	4.3	0.000	0.0	0.0	0.0	0.0	7607.	93967.
131-132	0.033	10.2	0.0	0.0	0.0	0.0	0.663	0.014	0.0	7607.	93967.
132-133	0.002	165.5	0.0	0.0	0.0	0.0	0.652	0.001	0.0	7607.	93967.

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Table D-1 Tug Geosynchronous Baseline Mission 3000 Lb Payload Delivery/Retrieval
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MISSION SEGMENT NO.	MISSION SEGMENT DURATION HR	MAIN ENG THRUST PROP. LBS	VENT PROP. LBS	TRANS-LATION PROP. LBS	ATTITUDE HOLD PROP. LBS	ATTITUDE MANEUVER PROP. LBS	STAB. CONTROL PROP. LBS	ROLL CONTROL PROP. LBS	THERMAL PROP. LBS	ROLL MOMENT SLUG-FT**2	YAW MOMENT SLUG-FT**2
133-134	0.033	0.0	0.0	0.0	0.010	0.0	0.0	0.0	0.0	7607.	92254.
134-135	0.767	0.0	0.0	0.0	0.222	0.0	0.0	0.0	0.3	7607.	92254.
135-136	0.083	0.0	0.0	0.0	0.0	1.226	0.0	0.0	0.0	7607.	92254.
136-137	0.050	0.0	0.0	0.0	0.015	0.0	0.0	0.0	0.0	7607.	92229.
137-138	0.083	0.0	0.0	0.0	0.024	0.0	0.0	0.0	0.0	7607.	92228.
138-139	0.083	0.0	0.0	0.0	0.0	1.226	0.0	0.0	0.0	7607.	92228.
139-140	0.014	0.0	0.0	35.9	0.004	0.0	0.0	0.0	0.0	7607.	92214.
140-141	1.000	0.0	0.0	0.0	0.291	0.0	0.0	0.0	0.4	7607.	91865.
141-142	0.014	0.0	0.0	35.7	0.004	0.0	0.0	0.0	0.0	7607.	91850.
142-143	0.050	0.0	0.0	0.0	0.015	0.0	0.0	0.0	0.0	7607.	91500.
143-144	0.050	0.0	0.0	0.0	0.015	0.0	0.0	0.0	0.0	7607.	91499.
144-145	0.450	0.0	0.0	0.0	0.131	0.0	0.0	0.0	0.2	7607.	91498.
145-146	0.450	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.2	0.	0.
146-147	0.333	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.1	0.	0.
147-148	0.021	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
148-149	0.053	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
149-150	0.039	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
150-151	0.062	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
151-152	0.042	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
152-153	15.000	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
153-154	0.133	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
154-155	0.567	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
155-156	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.	0.
TOTALS		53672.8	83.0	816.3	62.664	40.455	10.405	0.380	55.9		